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Journal of the
AMERICAN ROCKET SOCIETY
Rocketry . . . . Jet Propulsion Sciences . . . . Astronautics

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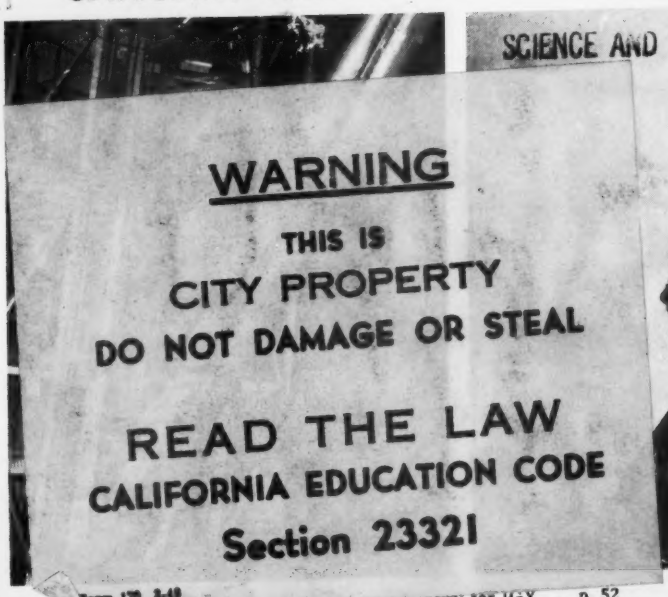

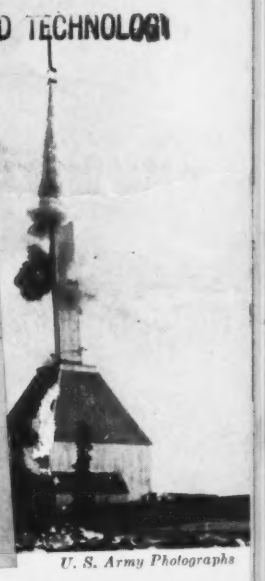

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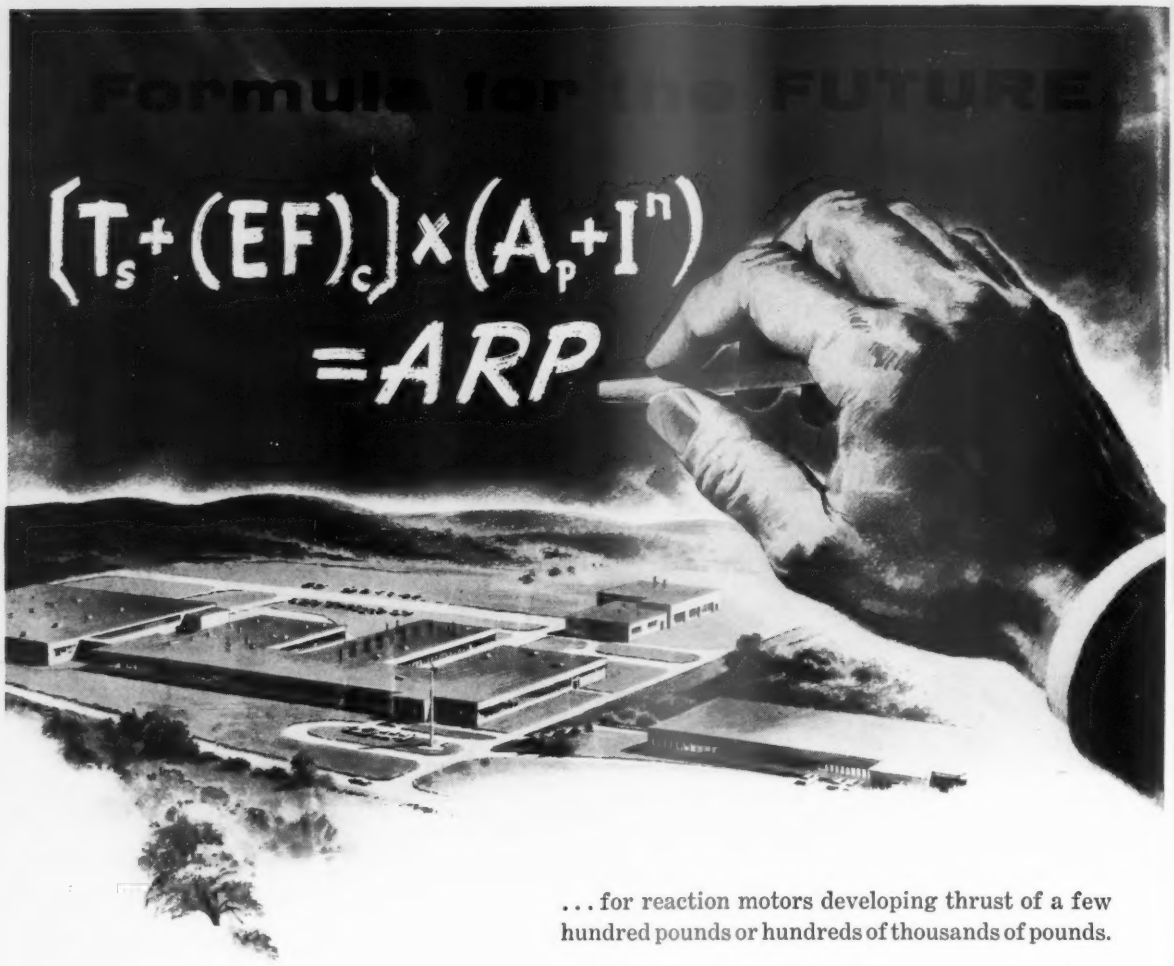

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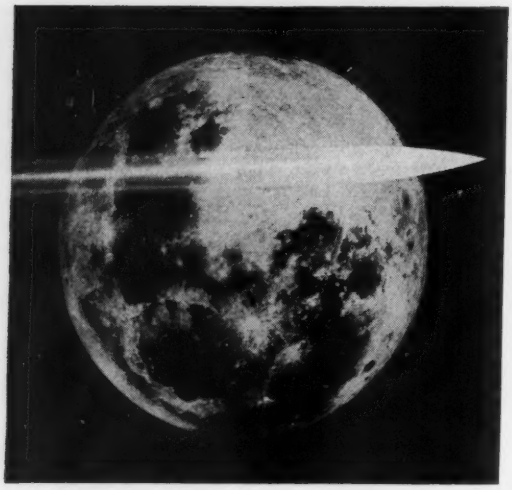

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JET PROPULSION, the Journal of the American Rocket Society, is devoted to the advancement of the field of jet propulsion through the publication of original papers disclosing new knowledge and new developments. The term "jet propulsion" as used herein is understood to embrace all engines that develop thrust by rearward discharge of a jet through a nozzle or duct; and thus it includes systems utilizing atmospheric air and underwater systems, as well as rocket engines. JET PROPULSION is open to contributions, either fundamental or applied, dealing with specialized aspects of jet and rocket propulsion, such as fuels and propellants, combustion, heat transfer, high temperature materials, mechanical design analyses, flight mechanics of jet-propelled vehicles, astronautics, and so forth. JET PROPULSION endeavors, also, to keep its subscribers informed of the affairs of the Society and of outstanding events in the rocket and jet propulsion field.

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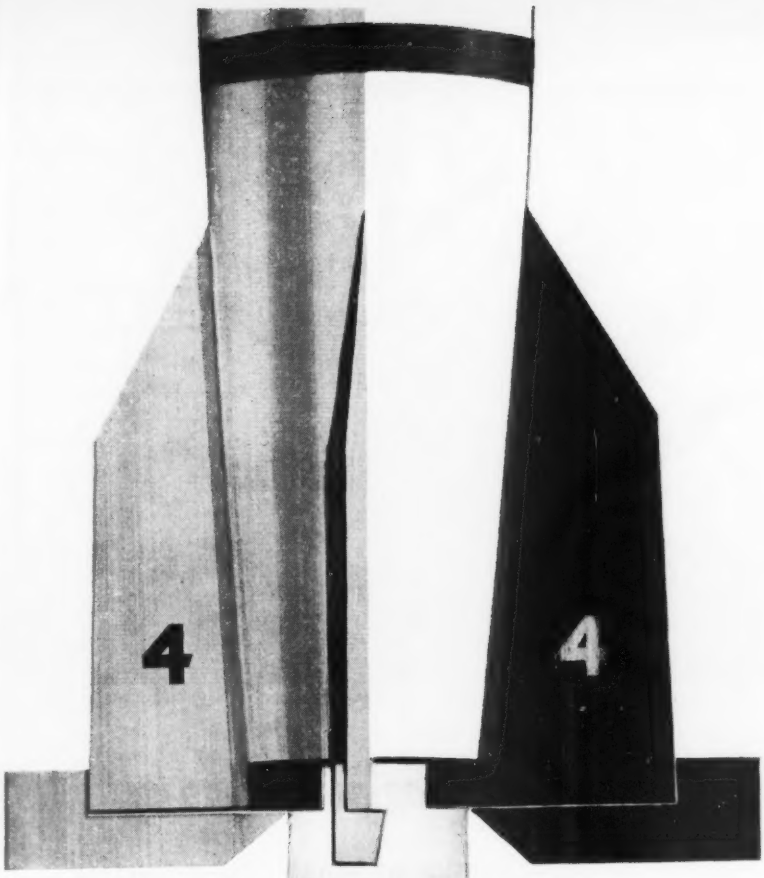

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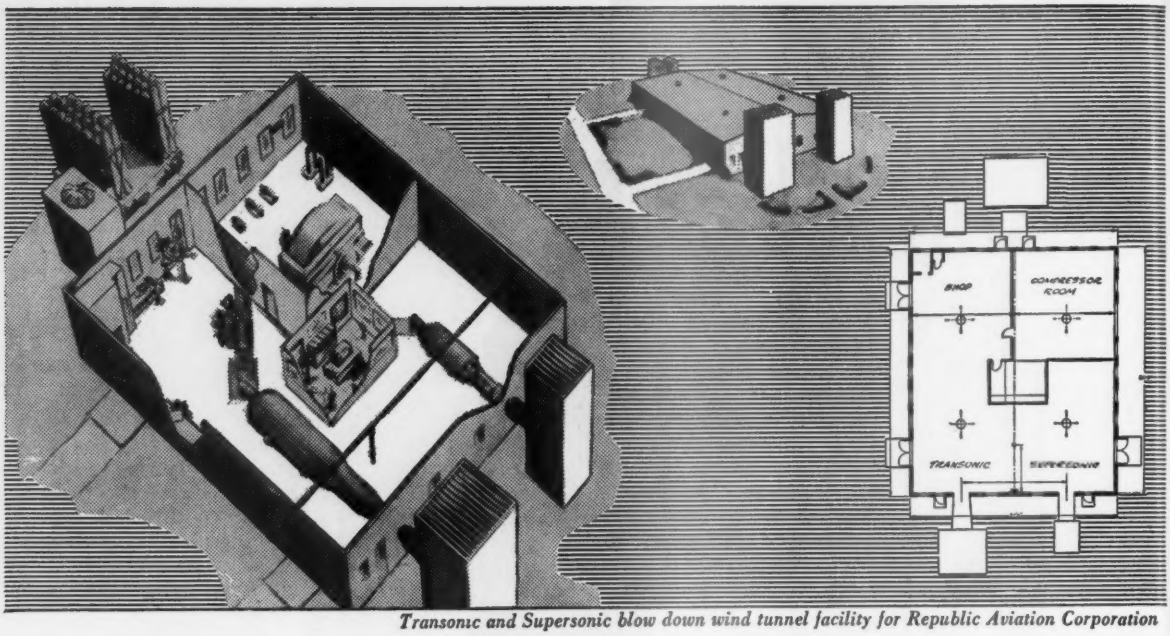

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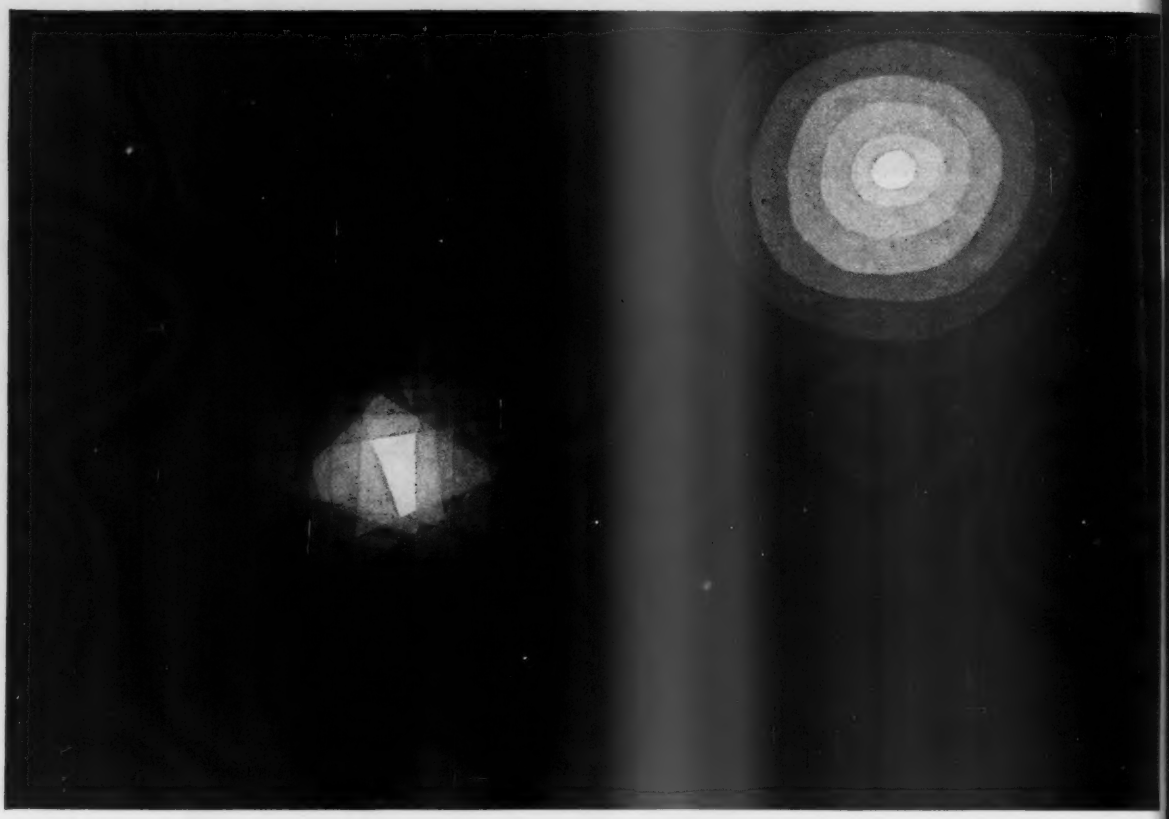

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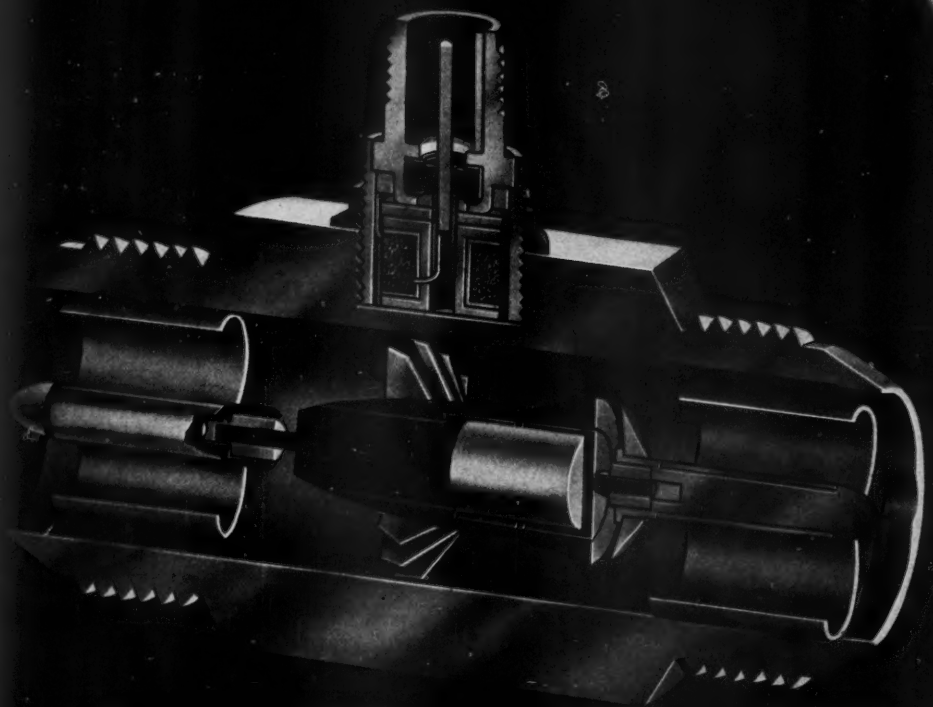

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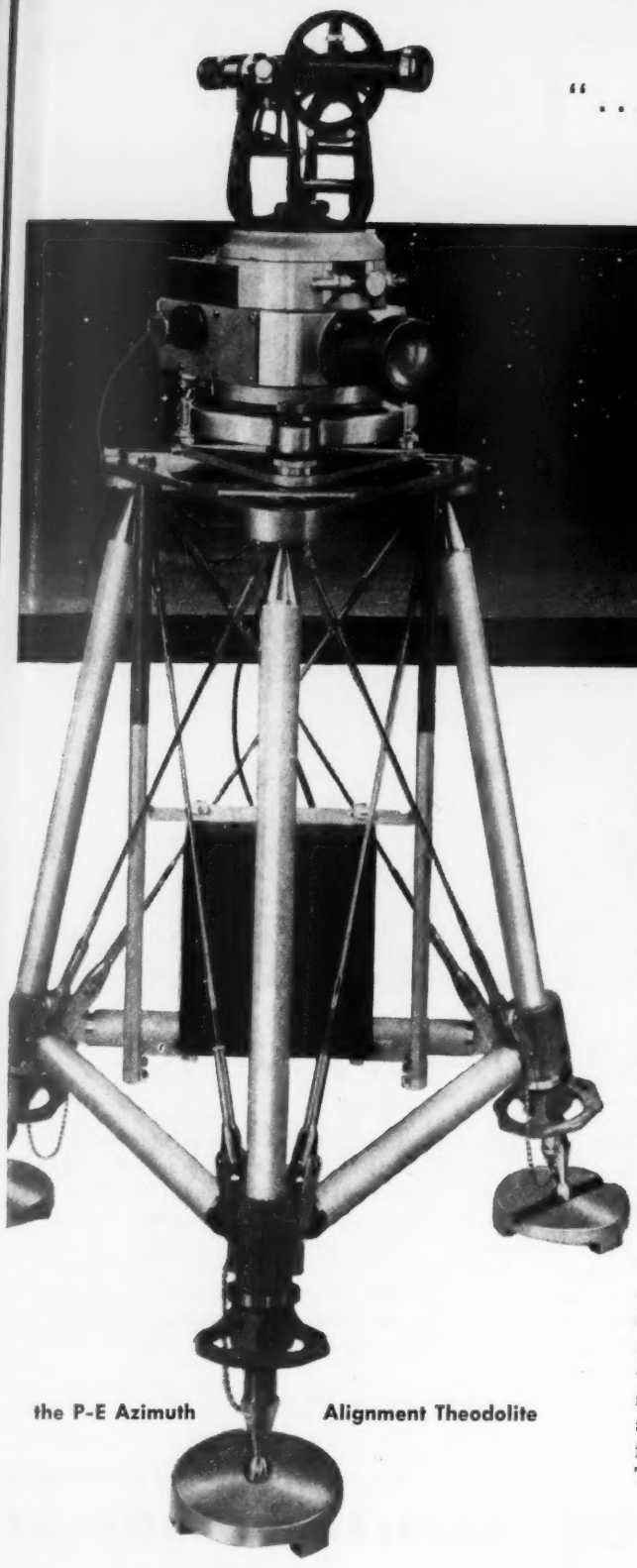

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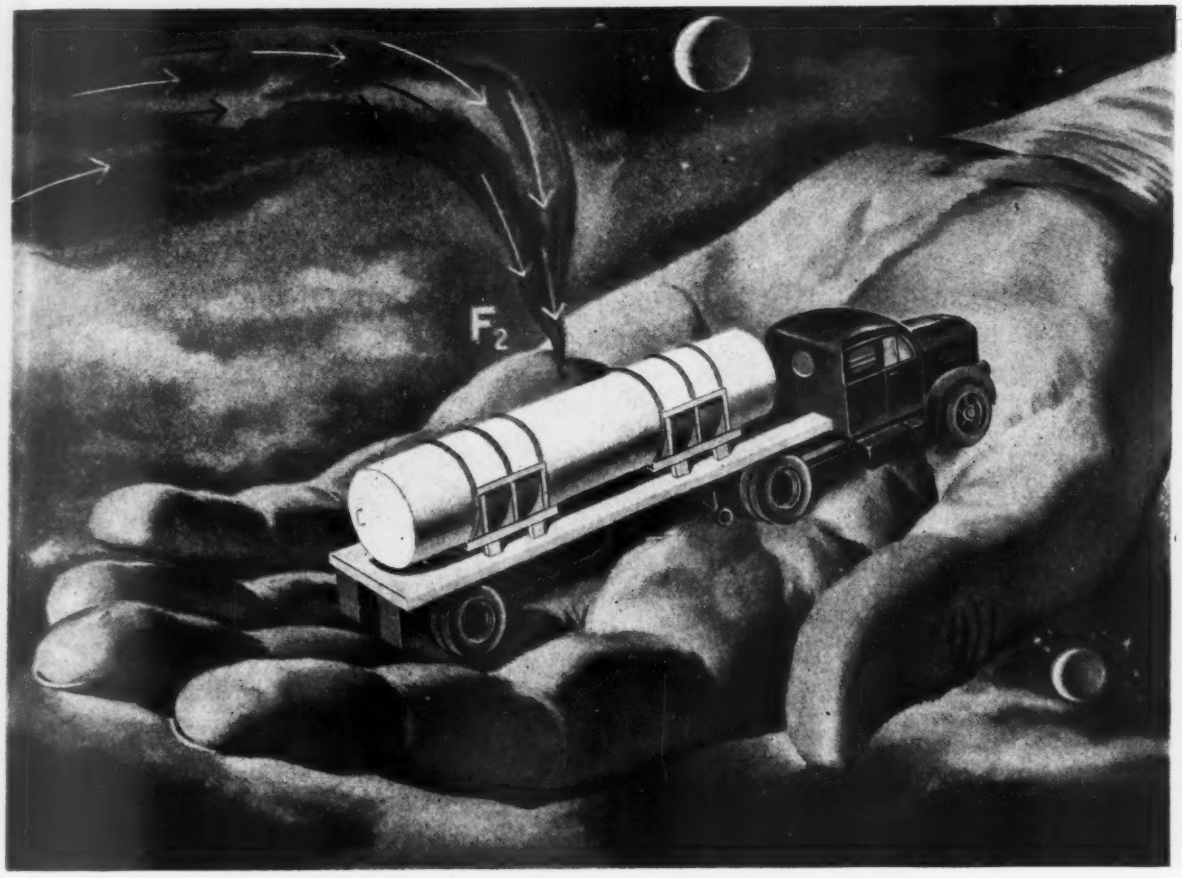

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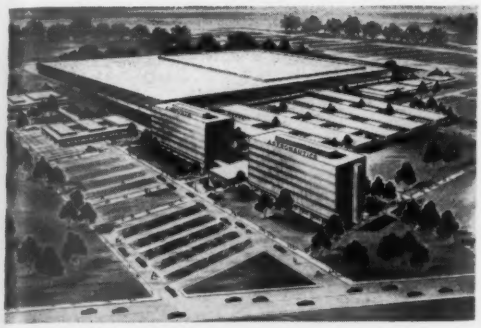

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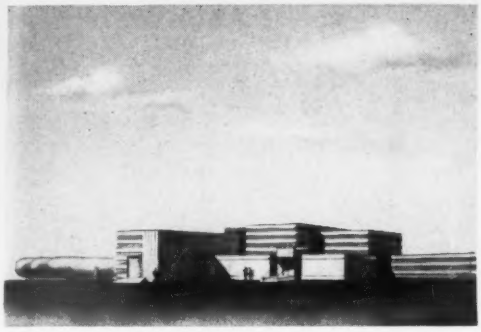

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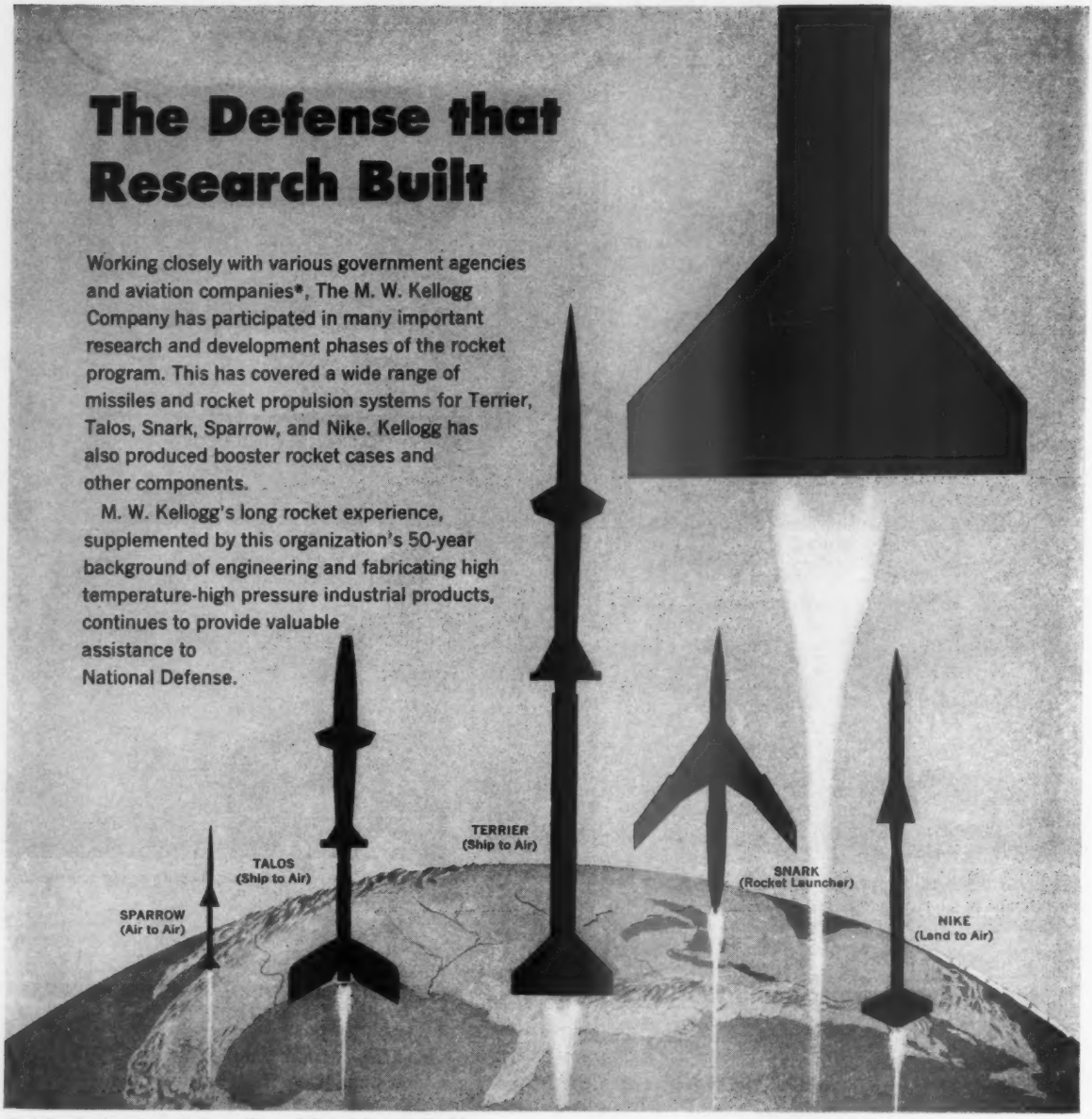

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Editor-in-Chief
MARTIN SUMMERFIELD

Lifetimes of Artificial Satellites of the Earth

IRVIN G. HENRY[1]

Aerojet-General Corp., Azusa, Calif.

The density of the upper atmosphere decays roughly exponentially with altitude. This variation is assumed to be exact, and the effect of the density is computed on the lifetimes of earth satellites in elliptical and in circular orbits. The physics of the drag is discussed.

Nomenclature

a = amplitude of variation in radius
A = frontal area of satellite
b = radius of satellite, assumed spherical
c = decay height of atmospheric mass density
C = a geometrical constant
C_D = drag coefficient
D = drag on the satellite
E = total (kinetic plus potential) energy of satellite
$g(r)$ = acceleration of gravity at radius r
G = acceleration of gravity at the surface of the earth = 9.81 meter/sec^2
h = height above the surface of the earth
H = solar heat received on earth per unit time per unit area = 1.35×10^6 erg/cm^2 − sec
k = Boltzmann constant = 1.38×10^{-16} ergs/°K
K = atmospheric constant
m = mass of an air atom
M = mass of the satellite
n = number of revolutions performed by the satellite
n_1 = number of revolutions performed by the satellite before its orbit becomes circular
n_2 = number of revolutions performed by the satellite before it strikes the earth
p = distance from the center of the earth to the perigee of the orbit
q = height of the perigee above the surface of the earth; $q = p - R$
r = distance from the center of the earth
r_0 = a constant
R = radius of the earth = 6370 km
T = temperature of the satellite
T = kinetic energy of the satellite
T_1, T_2 = hypothetical temperatures
v = velocity of the satellite
V = potential energy of the satellite
x, y = limits on n_2
z = dummy variable
α = radiative absorptivity of the satellite
β = radiative emissivity of the satellite
ϵ = eccentricity of the orbit
θ = azimuth angle around the earth
ρ = atmospheric mass density
σ = Stefan's constant = 5.67×10^{-5} erg/cm^2 − sec − °K^4

Presented at the ARS 11th Annual Meeting, New York, N. Y., Nov. 26–29, 1956.

[1] Physicist, Liquid Engine Division.

Introduction

ATTENTION has recently centered on the placing of small artificial satellites in orbits about the earth. These satellites would be retarded by the atmosphere and would ultimately spiral down and be destroyed. Their initial height is such that the drag is caused by individual molecular impacts and is calculable from kinetic theory. This article will attempt to estimate the lifetime of such a satellite.

The Drag Mechanism

If the satellite is close to the surface of the earth, its velocity is roughly $(GR)^{1/2}$, where G is the gravitational acceleration at the earth's surface and R is the radius of the earth; $(GR)^{1/2}$ is about 8×10^5 cm/sec. The root-mean-square velocity of the air molecules is $(3kT/m)^{1/2}$, where k is Boltzmann's constant, T is the absolute temperature, and m is the mass of a molecule. At very high altitudes the temperature might be as great as 1500 K and the air is largely monatomic, so that the molecular mass may be as low as 2.37×10^{-23} gm (14.5 on the atomic scale). This implies molecular velocities of order 1.5×10^5 cm/sec, which is much less than the velocity of the satellite. Hence it is a good approximation to calculate the drag of the satellite as if it were encountering molecules at rest.

Reflection of molecules from ordinary metal surfaces is generally diffuse rather than specular, and the molecules rebound with roughly the temperature of the metal surface. The satellite exchanges much more energy by radiative absorption and emission than by molecular impact, except during the last disastrous section of its flight. Therefore, the satellite's temperature is determined by radiation.

When the satellite is in the sun it receives $\pi b^2 H \alpha$ heat per unit time, where b is the radius of the satellite (assumed spherical), H is the solar flux density, and α is the radiative absorptivity at the wavelength of solar radiation (5000 Å). The satellite radiates $4\pi b^2 T^4 \beta \sigma$ heat per unit time, where β is the radiative emissivity at the wavelength corresponding to the temperature of the satellite (100,000 Å corresponds to 300 K). At equilibrium, the emitted power is equal to the absorbed power, giving $T = (H/4\sigma)^{1/4}(\alpha/\beta)^{1/4} = (\alpha/\beta)^{1/4} \times 280$ K. It is unlikely that $(\alpha/\beta)^{1/4}$ is greatly different from 1 for any common metal; if α and β referred to radiation of the same wavelength, then (α/β) would be exactly 1 by Kirchhoff's law.

When the satellite is in the earth's shadow it receives heat radiated from the earth. It is sufficiently close to the earth to consider the latter as a plane surface. When a particle is placed symmetrically between two infinite plane walls of temperature T_1 and T_2, it receives energy at the rate $(C/2)(T_1^4 + T_2^4)$ and emits at the rate CT^4, where C is some geometrical constant. Hence $T^4 = (1/2)(T_1^4 + T_2^4)$. If the particle is a

satellite, T_1 is the earth's temperature (roughly 300 K) and T_2 is the temperature of outer space (roughly 0 K). Therefore, the temperature of the satellite in the shadow approaches the equilibrium value $300/2^{1}/_{4} = 252$ K. At all events, the satellite tends to be only about 50 K cooler than the earth when it is in the shadow of the earth.

Thus, whether or not the satellite is in the shadow, the air molecules rebound from the surface of the satellite with a relative velocity corresponding to a temperature of 250 K or 300 K. This velocity is about 0.7×10^5 cm/sec, which is much less than the satellite's velocity of 8×10^5 cm/sec. Hence, in computing the drag D, it is a good approximation to say that in the coordinate frame in which the satellite is at rest, all the molecules approach with a uniform relative velocity of 8×10^5 cm/sec and rebound with zero relative velocity. If ρ is the atmospheric mass density, A is the satellite's frontal area, and v is the satellite's velocity, then the mass of molecules encountered by the satellite per unit time is ρAv and the change in momentum of the molecules encountered per unit time is ρAv^2. Thus the drag D of the satellite is ρAv^2; $D = (1/2)C_D\rho Av^2$, and the drag coefficient $C_D = 2$.

This is a well-known result for free molecule flow. The drag coefficient C_D can be somewhat greater than 2 if the molecules are not at rest before colliding with the satellite and if they leave the satellite with a noticeable velocity. Exact formulas for these more general cases are given by S. A. Schaaf and P. L. Chambré (3).[2] For near satellites of the earth, with the temperatures we have already mentioned, the exact drag coefficient appears to be about 2.04. When the satellite descends to about 140 km, the power received from molecular impacts becomes equal to the radiant power received from the sun, so the drag coefficient increases. When the satellite has descended to about 110 km, the mean free path becomes comparable with the diameter of the satellite, the individual molecular collisions are no longer independent, and the drag coefficient falls to the value which is associated with spheres in hypersonic (attached shock) continuum flow, which is about 0.7. These variations of the drag coefficient will not concern us, since our analysis of the motion will not follow the satellite below about 150 km. At this altitude the satellite will in any case perform only one or two more revolutions about the earth.

Atmospheric Density

For the purposes of this article we will asume that the atmospheric mass density is given by

$$\rho(h) = Ke^{-h/c} = Ke^{-(r-R)/c} \quad [1]$$

where h is the height above the surface of the earth, R is the radius of the earth, and K and c are constants. This functional relationship is quite close to the truth for values of h between 150 and 800 km. Experimental data have become available and tables of properties have been prepared by the Upper Atmosphere Rocket Research Panel (see 1, 2). The data above 160 km are still sketchy but indicate that we should take

$$K = 7.7 \times 10^{-10} \text{ gm/cm}^3$$
$$c = 23.3 \text{ km} = 2.33 \times 10^6 \text{ cm} \quad [2]$$

Description of the Motion

Usually the satellite will start life in an elliptical orbit. If the variation in height around the orbit is much greater than 23 km, the drag will have a sharp maximum at the perigee. The motion will then be very much as if the satellite suffered impulsive losses of velocity on each passage of the perigee. Thus the perigee will remain almost constant, while the apogee diminishes at each revolution (see Fig. 1). There will also be a slow rotation of the major axis, but this is a higher order

[2] Numbers in parentheses indicate References at end of paper.

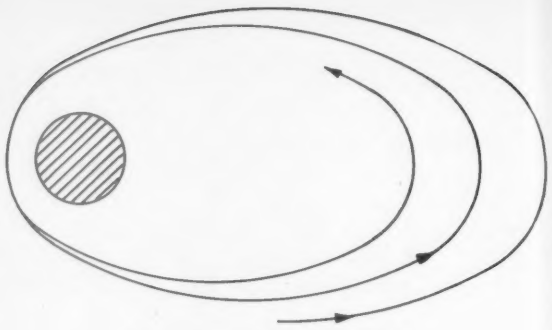

Fig. 1 Decay of elliptical orbit

effect which will not emerge from the simple analysis that we will present. Finally, the orbit will become almost circular and will assume the form of a spiral of gentle pitch. This form will persist until the satellite is destroyed in the lower atmosphere. Paradoxically, the effect of the drag will be to increase the mean velocity of the satellite. The form of the orbit is always close to that of a dragless orbit, and it is well known that the nearer a satellite is to its parent body, the higher is its speed. In descending, the satellite converts potential energy to kinetic energy. Only part of this potential energy is used to overcome drag; the remainder appears in the form of increased mean velocity.

Lifetime in a Circular Orbit

Let the satellite revolve with orbital velocity v near radius r. If the reduction in height during one revolution is very small, we can say that the loss in energy during one revolution ΔE is given by

$$\Delta E = -\oint Drd\theta = -\oint \rho(r)Av^2(r)rd\theta$$
$$= -AKe^{R/c}\int_0^{2\pi} e^{-r/c}v^2(r)rd\theta \quad [3]$$

The orbital velocity is found by remembering that

$$v^2(r)/r = g(r)$$

where $g(r)$ is the gravitational acceleration at radius r. Hence

$$v^2(r) = GR(R/r) = GR^2/r$$

where $(GR)^{1/2}$ is the circular orbital velocity at R, and G is the gravitational acceleration at R; i.e., G is 9.81 meter/sec^2. Thus

$$\Delta E = -AKGR^2 e^{R/c}\int_0^{2\pi} e^{-r/c}d\theta$$
$$= -2\pi AKGR^2 e^{R/c}e^{-r/c} \quad [4]$$

The decrement in radius Δr, corresponding to the decrement in energy ΔE, is given by

$$\Delta r = 2\Delta E/Mg(r) = 2(\Delta E/MG)(r/R)^2 \quad [5]$$

where M is the mass of the satellite. The factor 2 enters because, in descending a distance Δr, the satellite loses an amount $Mg(r)\Delta r$ of potential energy; but, because closer orbits are faster, it gains $(1/2)Mg(r)\Delta r$ in kinetic energy. Equating the two expressions for ΔE in [4, 5] ,we find

$$\Delta r = -4\pi(A/M)Ke^{R/c}r^2e^{-r/c} \quad [6]$$

If Δr is very small, we can treat it as a true differential to find the long-term mean motion of the satellite

$$r^{-2}e^{r/c}dr = -4\pi(A/M)Ke^{R/c}dn$$

Substituting $r = R + h$, we find

$$(R + h)^{-2}e^{h/c}dh = -4\pi(A/M)Kdn$$

Expanding the left member by the binomial theorem and integrating

$$e^{h/c}[c - (2/R)(ch - c^2) + \ldots]_{h_1}^{h_2} = -4\pi R^2(A/M)K(n_2 - n_1)$$

If the initial height h_1 is q and if the final height h_2 is 0 (ground level), and if $q \gg c$, we have

$$(n_2 - n_1) = (1/4\pi)(c/R^2K)(M/A)e^{q/c}[1 - (2q/R)] \ldots [7]$$

Substituting the values already given for K and c, and remembering that $2\pi R = 4 \times 10^9$ cm (the original definition of the centimeter), we obtain

$$n_2 - n_1 = 5.95 \times 10^{-4}(M/A)e^{q/c}[1 - (2q/R)] \ldots . [8]$$

where M/A, the mass per frontal area of the satellite, is given in gm/cm². A plot of this function is presented in Fig. 2. It will be seen that an increase of 150 km in starting altitude multiplies the lifetime by about 10^3. Thus it is unreasonable to lengthen the life of a satellite with a small long-duration rocket motor, since a slight increase in altitude will accomplish the same result.

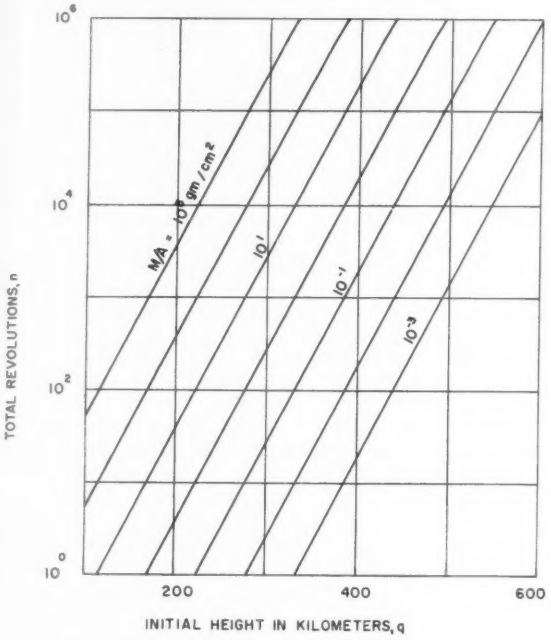

Fig. 2 Lifetime of a satellite in a circular orbit

Lifetime in an Elliptical Orbit

If there is no atmospheric drag, the satellite travels in an ellipse of the form

$$1/r = (1/r_0)(1 + \epsilon \cos\theta)$$

$$r = r_0(1 - \epsilon \cos\theta + \epsilon^2 \cos^2\theta - \ldots)$$

where r_0 and ϵ are constants. We will approximate this form by writing

$$r \cong p + a(1 - \cos\theta) \ldots [9]$$

Thus p is the distance from the center of the earth to the perigee; $p + 2a$ is the distance from the center of the earth to the apogee; and a is the amplitude of variation in radius (see Fig. 3). This approximation for r will be satisfactory as long as a is less than or not much greater than p. We can write the energy decrement per turn just as we did for circular orbits

$$\Delta E = -\oint Dvdt \cong -\oint Dpd\theta$$

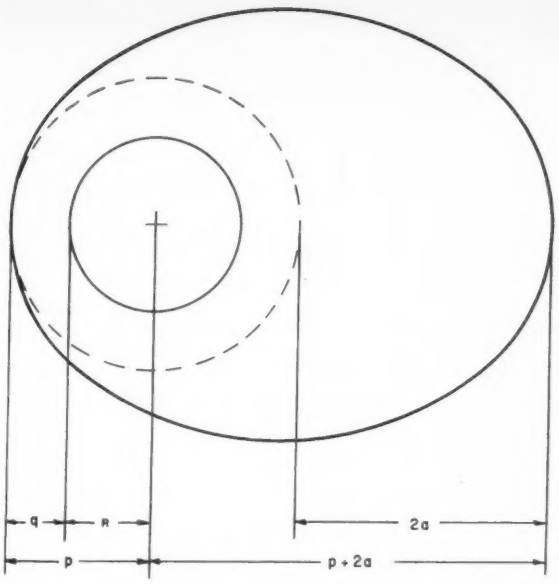

Fig. 3 Illustration of symbols

In general $(vdt)^2 = (rd\theta)^2 + (dr)^2$. However, the drag is very strongly peaked around the perigee, where r is passing through a minimum. Hence we can ignore dr compared with $rd\theta$. The fact that the drag has such a strong maximum also allows us to substitute $pd\theta$ for $rd\theta$. We have the relation for the drag

$$D = KAe^{-(r-R)/c}v^2 \ldots [10]$$

so

$$dD/D = -dr/c + 2dv/v \ldots [11]$$

But dv/v is of order dr/r which we have already neglected, whereas dr/c is r/c times greater, i.e., about 10^3 times greater. Hence in writing D we have to retain the θ-variation of r in the exponential, while dropping the θ-variation of v^2.

The velocity at p can be found in the following way: The major axis of the orbit is $2(p + a)$; see Fig. 3. Hence the total energy E of the satellite is $-MGR^2/2(p + a)$. However the potential energy $V(p)$ at radius p is $-MGR^2/p$. Hence the kinetic energy $T(p)$ at radius p is given by

$$T(p) = E - V(p)$$

$$= MGR^2(p + 2a)/2p(p + a)$$

so

$$v^2(p) = GR^2(p + 2a)/p(p + a)$$

and

$$D = KAGR^2e^{R/c}e^{-(p+a)/c}e^{(a/c)\cos\theta}(p + 2a)/p(p + a)$$

Inserting this result into ΔE, we obtain

$$\Delta E = -KAGR^2e^{R/c}e^{-(p+a)/c} \times [(p + 2a)/(p + a)] \int_0^{2\pi} e^{(a/c)\cos\theta} d\theta$$

$$= -2\pi KAGR^2e^{R/c}e^{-(p+a)/c}[(p + 2a)/(p + a)]I_0(a/c). [12]$$

where $I_0(a/c)$ is the modified Bessel function of the first kind, of zero order, and of argument a/c. We substitute for this function an expansion valid when $a/c \gg 1$, and find

$$\Delta E = -\sqrt{2\pi}\, KAGR^2e^{R/c}e^{-p/c}[(p + 2a)/(p + a)](c/a)^{1/2}. [13]$$

That the expansion introduces a very small error we demonstrate in Fig. 4, where we show

$$I_0(x) \Big/ [e^x/\sqrt{2\pi x}]$$

Now

$$E = -[MGR^2/2(p + a)] \ldots [14]$$

and p remains almost unchanged when the drag is concentrated

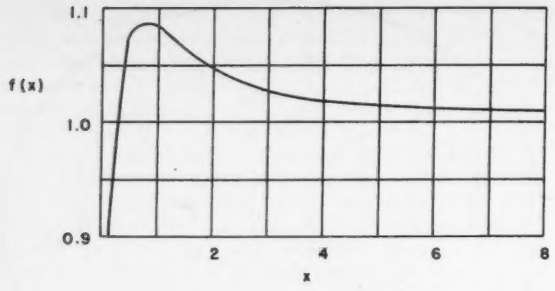

Fig. 4 $f(x) = I_0(x)/\left[e^x/\sqrt{2\pi x}\right]$

near the perigee, i.e., when $a/c \gg 1$, so that

$$\Delta E = [MGR^2/2(p+a)^2]\Delta a \quad \ldots\ldots [15]$$

Equating the last two expressions for ΔE, we obtain for the loss in amplitude per turn

$$\Delta a = -2\sqrt{2\pi}\,K(A/M)e^{R/c}e^{-p/c}(p+a)(p+2a)\sqrt{c/a} \quad \ldots [16]$$

Considering many revolutions of the satellite, we can assume that a varies smoothly and we can treat Δa as a true differential

$$\frac{\sqrt{a}\,da}{(p+2a)(p+a)} = -2\sqrt{2\pi}\,K(A/M)e^{-(p-R)/c}\sqrt{c}\,dn$$

If we let $a = z^2$, the left member becomes

$$\frac{2z^2\,dz}{(p+2z^2)(p+z^2)} = 2dz\left[\frac{1}{p+z^2} - \frac{1}{p+2z^2}\right]$$

This can be integrated to give

$$\frac{2}{\sqrt{p}}\arctan\frac{z}{\sqrt{p}} - \frac{2}{\sqrt{2p}}\arctan\sqrt{\frac{2}{p}}\,z = \frac{2}{\sqrt{p}}\arctan\sqrt{\frac{a}{p}} - \sqrt{\frac{2}{p}}\arctan\sqrt{\frac{2a}{p}}$$

When the amplitude is a, $n = 0$; and when the amplitude is 0, $n = n_1$. Thus the number of revolutions performed before the ellipse decays to a circle is

$$n_1 = \frac{1}{\sqrt{2\pi}}\frac{1}{Kc}\frac{M}{A}\sqrt{\frac{c}{p}}\,e^{q/c} \times \left[\arctan\sqrt{\frac{a}{p}} - \frac{1}{\sqrt{2}}\arctan\sqrt{\frac{2a}{p}}\right]$$

$$= 222\,\frac{M}{A}\sqrt{\frac{c}{p}}\,e^{q/c} \times \left[\arctan\sqrt{\frac{a}{p}} - \frac{1}{\sqrt{2}}\arctan\sqrt{\frac{2a}{p}}\right] \quad \ldots\ldots [17]$$

where M/A is in gm/cm². The function in square brackets is plotted in Fig. 5. If a is much smaller than p, which will be the usual case, we can expand the arctangents to obtain

$$n_1 = [1/3(2\pi)^{1/2}](1/Kc)(M/A)(c/p)^{1/2}e^{q/c}(a/p)^{3/2}$$

$$= 74.0\,(M/A)(c/p)^{1/2}e^{q/c}(a/p)^{3/2} \quad \ldots\ldots\ldots\ldots [18]$$

where M/A is given in gm/cm².

Maximizing the Total Life

The total number of turns executed is thus approximately

$$n_2 = (1/4\pi)(c/R^2K)(M/A)e^{(p-R)/c}(R/p)^2 [1 + (2\sqrt{2\pi}/3)(a/c)^{3/2}] \quad \ldots\ldots [19]$$

where the first term is the contribution of the circular orbit and the second is that of the elliptical orbit. In general $p/R \sim 1$ while $a/c \gg 1$, so that the satellite will usually spend the

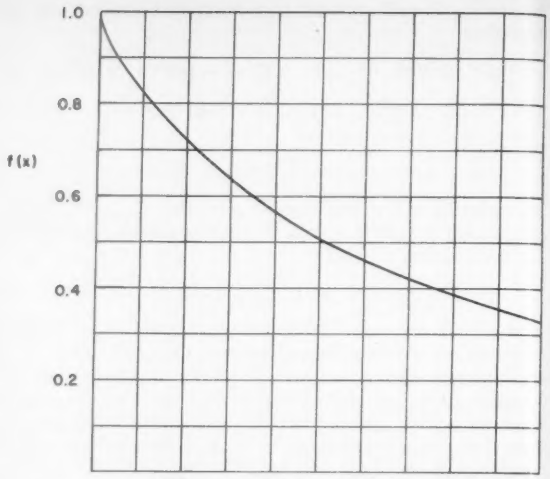

Fig. 5 $f(x) = \dfrac{\arctan x^{1/2} - 2^{-1/2}\arctan(2x)^{1/2}}{(x^{3/2}/3)}$

greater part of its life in the elliptical orbit. However, the usual situation is that a given energy is available for launching the satellite, and this implies that a fixed $(p + a)$ is attainable. The term $e^{p/c}$ completely outweighs the term $(a/c)^{3/2}$, so that if maximum life is required from the satellite, then a should be as small as possible while p is made as large as possible; that is, the satellite should be launched in a circular orbit. For instance, let us compare two satellites, both of which have $M/A = 10$ gm/cm². Let the first satellite have $a = 500$ km, $q = 500$ km, $p = R + q = 6870$ km; and let the second satellite have $a = 0$, $q = 1000$ km, $p = 7370$ km. Thus both satellites have the same total energy. However, the first will perform 2×10^9 elliptical turns followed by 10^7 circular turns, while the second will perform 10^{16} circular turns.

Usually a further complication arises because a and p cannot be exactly specified, since the guidance and propulsion systems have a limited accuracy. The result is that unless p and a can be predetermined with an accuracy $\pm c/2$, it is impossible to determine the lifetime within a factor e. This makes it difficult to design certain experiments, particularly those in which the atmospheric mass density is deduced from the decay of the orbit. Consequently (see Fig. 6), a rather

(*Continued on page 27*)

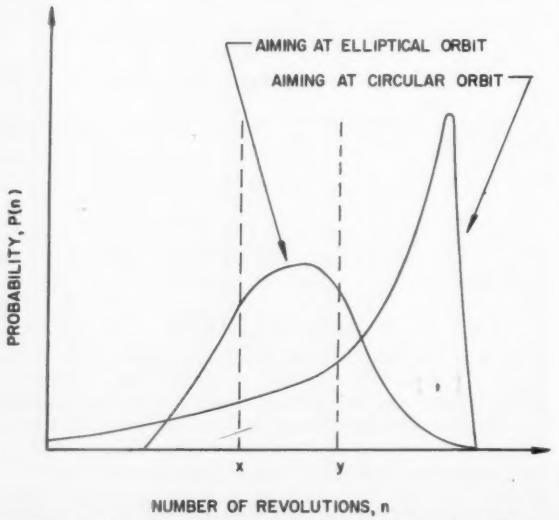

Fig. 6 Maximizing the useful part of the life of a satellite

Effect of Turbojet Compressor Coolants on Afterburner Combustion

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In the combined use of compressor coolant injection and afterburning as a means of achieving maximum thrust augmentation of a turbojet engine, the coolants investigated (water-alcohol mixture and liquid ammonia) reduced the combustion temperature and efficiency, and limits of stable combustion in the afterburner. The mechanism of this negative influence is related to the effect of water vapor and ammonia on the combustion propagation mechanism.

Introduction

IN THE continuing effort to increase the thrust output of the turbojet engine, several methods of thrust augmentation have been investigated both singly and in combination. The most common and most productive method of thrust augmentation is *afterburning*. Afterburning involves the addition of heat during the exhaust portion of the turbojet engine cycle by burning additional fuel downstream of the turbine in the turbojet tailpipe. More moderate thrust increases can be obtained by *compressor coolant injection*, another relatively common method of thrust augmentation. Injection of a coolant ahead of or during the compression process lowers the temperature, thus increasing the air density and compressor-pressure ratio.

To obtain the ultimate in thrust augmentation, afterburning has been combined with compressor injection. A schematic diagram of the *combined augmentation system* is shown in Fig. 1. Cycle analyses indicate that compressor coolant injection should directly augment the thrust of the afterburner in a predictable manner (1).[2] However, it has been found experimentally that the presence of the coolant so changes the combustion and reaction characteristics of the gases that the afterburner combustion performance is deteriorated to such an extent that the predicted agumentation is not achieved.

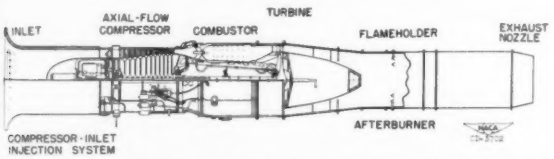

Fig. 1 Schematic diagram of turbojet engine with compressor inlet injection and afterburner augmentation system

During the course of turbojet thrust augmentation investigations conducted at the NACA Lewis laboratory, the effects of two coolants, a water-alcohol mixture and liquid ammonia, on the afterburner performance were determined, and are summarized herein. Also, an effort is made to explain the observed decrement in combustion temperature, efficiency, and stability limits on the basis of fundamental combustion characteristics associated with the environmental changes imposed by the coolants.

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Effect of Water Vapor

The ready availability and high heat of vaporization of water made it a logical choice as a compressor coolant in the early studies of compressor coolants. Alcohol was added to prevent freezing at the moderatly low temperatures encountered at sea level during winter. A coolant composition of 70 per cent water, 15 per cent methyl alcohol, and 15 per cent ethyl alcohol by volume was used.

Additional heat would normally be required to raise the temperature of the coolant to that of the combustion in the engine combustor. However, it was found that the combustion of the alcohol in the water-alcohol mixture would provide the additional heat necessary for raising the temperature of the water vapor. Thus, additional fuel would not be required and a constant engine throttle setting could be used during the coolant injection. This latter condition would simplify the engine-control and fuel-system design and operation.

The water-alcohol mixture is completely vaporized after passing through the turbine before it passes through the afterburner. Thus the coolant is thoroughly mixed with the gases and fuel entering the afterburner.

Fig. 2 shows the experimentally determined afterburner combustion temperatures that demonstrate the effect of increasing the coolant-to-air ratio for a range of over-all equivalence ratios. Over-all equivalence ratio is the quotient of the actual combustible-to-air ratio in the engine and afterburner to the stoichiometric mixture ratio. This term includes the engine fuel, the afterburner fuel, and the portion of the coolant that is combustible.

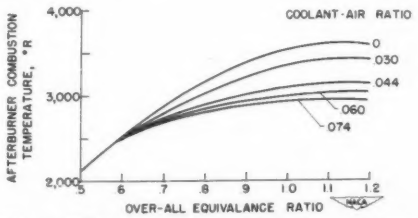

Fig. 2 Effect of compressor injection of 70 per cent water–30 per cent alcohol mixture on afterburner combustion temperature

When a 70 per cent water–30 per cent alcohol mixture is injected at the compressor inlet with the afterburner engine combination operating stoichiometrically (equivalence ratio of 1.0), increasing the coolant-air ratio from 0 (afterburning alone) to 0.074 produces a decrease of 850 R in the afterburner-combustion temperature, as is shown in Fig. 2. As the equivalence ratio is decreased from 1.0, similar water-air ratios cause a smaller decrease in combustion temperature, until at an equivalence ratio of 0.6 the addition of the coolant has a negligible effect on the combustion temperature. The decrease in combustion temperature is reflected in a corresponding decrease in combustion efficiency, as is shown in Fig. 3.

An explanation of the phenomena involved in the influence of water-based coolants on the combustion process in the

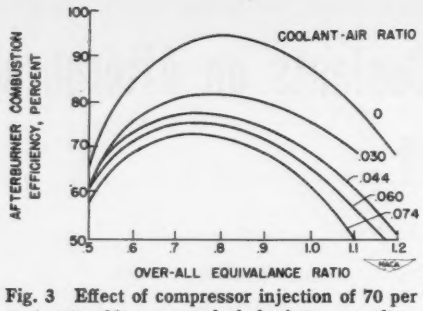

Fig. 3 Effect of compressor injection of 70 per cent water–30 per cent alcohol mixture on afterburner combustion efficiency

afterburner can be shown to lie in the effect of water vapor on the mechanism of combustion propagation. The advancement of a flame front in a combustible mixture is assumed to be normally accomplished by thermal convection and to some extent by radiation from layer to layer of the fuel-air mixture. The water vapor introduced into the combustion zone from the compressor coolant would reduce both the thermal convection and radiation, and would absorb some of the free radicals present within the flame front. The result would be a reduction of the flame temperature. Wohl and von Elbe (2) and Garner and Johnson (3, 4) have demonstrated in some early work that the water vapor has a quenching effect on radiation involved in the flame front propagation. Brokaw and Gerstein (5) have shown recently that the burning velocity, quenching distance, and minimum ignition energies of a hydrocarbon flame are adversely affected by any reduction in the flame temperature, such as would be caused by the addition of water vapor.

Efforts to quantitatively correlate the influence of water vapor on flame speed and reaction rate of hydrocarbon flames with the observed effect of water on the combustion in an afterburner have not proved successful to date. However, a qualitative comparison is possible. When small concentrations of water vapor are added to stoichiometric mixtures in small-scale laboratory experiments, there is only a small effect on the flame speed, quenching distance, and reaction rate of the mixture. As the quantity of water vapor is increased, these fundamental combustion parameters are more adversely affected. The full-scale engine data presented herein also show the same result. Thus, although the actual mechanism of the losses is still questionable, there is no doubt that it involves the flame speed and reaction rate.

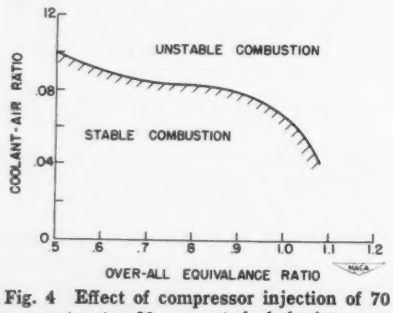

Fig. 4 Effect of compressor injection of 70 per cent water–30 per cent alcohol mixture on afterburner operational limits

It might be suspected that if the combustion efficiency and temperature are affected by the water-alcohol mixture, then the limits of stable combustion will also be affected. Fig. 4 shows the region of stable combustion in the afterburner as the equivalence ratio and the coolant-air ratio are varied. Operation of the afterburner was impossible above the indicated operational limit. The operational limit shown here was imposed by screeching combustion rather than by combustion blowout, which would have occurred at a higher equivalence ratio. The screeching combustion limit cannot be considered a general limitation, but rather as being related to the particular afterburner design used. Unfortunately, in these tests screeching combustion was not controllable and the actual rich combustion limit could not be determined.

Effect of Ammonia Vapor

The use of a compressor coolant during subsonic flight at high altitudes requires the selection of another coolant with a freezing temperature lower than that of water. The static temperature at an altitude of 35,000 ft is −66 F, and with a flight Mach number of 1.0 the temperature is increased only to 13 F. In the selection of a suitable coolant for altitude use, a compromise between vapor pressure and heat of vaporization would be the use of liquid, anhydrous ammonia. Liquid ammonia freezes at −27 F at atmospheric pressure and has a heat of vaporization of 561 Btu/lb, about half that of water.

Experimental investigations with a full-scale afterburner have shown that ammonia vapor also has an effect on the afterburner combustion, but an effect of a different nature than that produced by water vapor. Fig. 5 shows that the

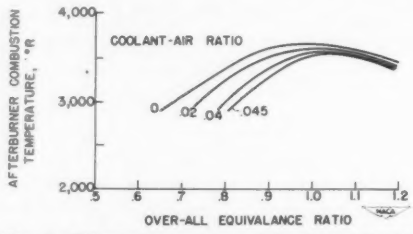

Fig. 5 Effect of compressor injection of liquid, anhydrous ammonia on afterburner combustion temperature

combustion temperature decrease is rather small at stoichiometric and that as the equivalence ratio is reduced below 0.9 the combustion temperature decreases more rapidly, especially as the coolant-air ratio is increased. Accompanying this decrease in combustion temperature is a similar decrease in combustion efficiency (Fig. 6), when the coolant-air ratio is increased at equivalence ratios below 0.9. Varying the ammonia concentration had no effect on the combustion efficiency of a stoichiometric mixture.

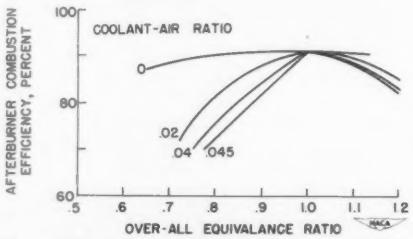

Fig. 6 Effect of compressor injection of liquid, anhydrous ammonia on afterburner combustion efficiency

Very low concentrations of ammonia have the property of shortening the induction or pre-ignition period of the combustion process of hydrocarbon flames. As the ammonia concentration is increased above a certain point, further increases will inhibit the initiation and growth of active energy carriers that are in part responsible for the propagation of the flame front. This effect was demonstrated by Williamson and Pickles and is reported in (6). Taylor and Salley (7) showed that there exists a concentration of ammonia for which the combustion reaction rate is a maximum and further in-

creases in ammonia concentration will have an adverse effect on the flame propagation and thus on the combustion process. The inhibitive effects become appreciable when the concentration of ammonia reaches values greater than 1 per cent. The data shown in Figs. 5 and 6 had a minimum concentration of 2 per cent and, therefore, operation was in the region where increasing the ammonia concentration tended to inhibit the flame propagation.

As the combustion temperature is increased above 3500 R (equivalence ratios greater than 0.9) the ammonia is dissociated into radicals that exhibit a different effect on flame propagation from that of the ammonia itself. The radicals formed from the dissociation of ammonia do not reduce the combustion reaction rate as did the ammonia molecules. Therefore, the combustion temperature and efficiency (Figs. 5 and 6) were not significantly affected during stoichiometric operation.

The ammonia vapor had an effect on the region of stable combustion in the afterburner, as is shown in Fig. 7. Operation was possible only at coolant-air ratios below 0.045 for equivalence ratios below stoichiometric and with a maximum of 0.045 with stoichiometric combustion. The stability limitation in this case was actual cessation of combustion or blowout of the afterburner. As the equivalence ratio was decreased the maximum coolant-air ratio that permitted operation tended to decrease until at an equivalence ratio of 0.5 the operable coolant-air ratio was less than 0.02.

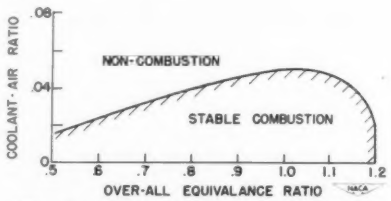

Fig. 7 Effect of compressor injection of liquid, anhydrous ammonia on afterburner combustion stability limits

The operational limit shown in Fig. 7 is a function of many factors including the rich limit of combustion, the fuel-air distribution in the burning zone, and the flame stabilization configuration. For this reason the actual limit indicated may be a function of the particular geometry employed. The trends indicated, however, are believed to be typical.

Concluding Remarks

The coolants evaporated during compressor injection have a deleterious influence on the combustion performance in the afterburner used in a combined augmentation system. When a water-alcohol mixture is used as the coolant, the combustion temperature and efficiency are progressively decreased as the coolant-air ratio is increased. The effect of the water-base coolant is less at lower equivalence ratios than at stoichiometric. Although the detailed mechanism of the losses is still questionable, the role of the flame speed and reaction rate have been recognized.

When liquid ammonia is used as the compressor coolant, the combustion performance is also reduced, but the nature of the effect is different from that of water. The greatest decrease in combustion efficiency and temperature was obtained when the ammonia-air ratio was increased and the afterburner was operating at equivalence ratios below 0.9. Increasing the ammonia concentration had no effect on the combustion of a stoichiometric mixture. The high temperature associated with stoichiometric combustion dissociated the ammonia into radicals that do not exhibit an inhibitive effect on the flame propagation with the result that the ammonia had only a small influence on the stoichiometric combustion.

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Lifetimes of Artificial Satellites of the Earth

(*Continued from page 24*)

unusual optimization problem occurs. If one uses the energy to aim at the longest life (i.e., circular orbit), one obtains a peaked probability distribution with a large peak very close to the maximum possible theoretical life. There is also a long tail which may extend to lives so short that the mission is a failure. If one aims at a shorter life, maintaining the same $(p + a)$ but increasing a, the peak becomes broader and occurs at a lower lifetime. The problem is then to maximize

$$\int_x^y P(n_2)dn_2$$

where y and x are the upper and lower limits of n_2, which would still result in a useable experiment.

Applications of the Results

The treatment used to find the lifetime in a circular orbit is accurate; the expression for the lifetime in elliptical orbits is less accurate, the major error being caused by the assumption that the perigee does not diminish at all. This assumption yields a slight overestimation of the lifetime. The expression given for the atmospheric mass density is merely an estimate when applied above 300 km and may be incorrect by a factor of 10 at 800 km. This would result in an error of 10 in the lifetime of a satellite with a perigee height q of 800 km. This is inconsequential, since the lifetime of any satellite at that altitude is infinite for all practical purposes. Finally we would like to point out that, even if the density does not vary rigorously with $e^{-h/c}$, the preceding treatment is still applicable. In this event K becomes $K(r)$ or $K(a)$, and for any reasonable $K(r)$ the expression obtained for dn/dr can still be integrated. The final results for n_1 and n_2 will of course be slightly changed.

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Flame Stabilization in a Boundary Layer

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The theory of flame stabilization is examined by studying the stability of a lean propane-air flame in the boundary layer along a water-cooled slender rod, with its longitudinal axis lying at the center line of a large Pyrex duct. The extraneous dilution effect due to surrounding atmosphere present in the study of the stability of a Bunsen flame is thus eliminated. The position of the flame from the leading edge of the slender rod is found to be independent of the rod position inside the duct and of the duct length, unless the flame is stabilized within 2 in. of the duct exit. This indicates that, except in the latter region, the stable flame position is determined by the flow characteristics in the boundary layer along the rod. The distance between the flame and the leading edge of the rod decreases as the free stream mixture velocity decreases, the rod coolant temperature increases, and the propane-air ratio increases from lean toward the stoichiometric value. For a given coolant temperature, the velocity gradient at the rod surface, calculated at the stable flame positions by assuming a Blasius velocity profile in the boundary layer, is correlated with the propane-air ratio. The experimental scatter is approximately ±10 per cent.

Nomenclature

F = propane-air ratio
p = pressure
R_x = length Reynolds number, $u_\infty x/\nu_\infty$
T = temperature
u = velocity component in x-direction
x = distance along the stabilizer surface from the leading edge
y = distance normal to the stabilizer surface
ν = kinematic viscosity

Subscripts

∞ = free stream conditions
c = stabilizer coolant
f = flame
s = stoichiometric mixture
w = condition at stabilizer wall

Introduction

THE theory of flame stabilization has often been examined by studying experimentally the stability of a Bunsen flame at two extreme conditions; viz., flashback and blowoff. The mixture ratio at which each of these limits occurs has been found to depend substantially uniquely on the boundary velocity gradient. See, for example, (1, 2).[4]

Some difficulties are present, however, in the stability study of a Bunsen flame. These include dilution due to surrounding atmosphere and great divergence of flow near the flame fringe, where the conditions are important in determining flame stability. Because of these extraneous effects, errors are introduced in the calculation of the stability limits by the usual method. Furthermore, the criterion for stability can only be studied under the two extreme conditions of flashback and blowoff. These conditions are on the verge of instability, with the flashback limit controlled by the cooling of the burner wall and the blowoff limit controlled by the dilution due to the surrounding atmosphere. Although the flames are stable near the burner exit between these two limits, the stability criterion cannot be studied there because the burning-velocity profile is not known.

In order to eliminate the difficulties mentioned above, the theory of flame stabilization is re-examined by studying the stability of a flame in the boundary layer along a water-cooled slender rod, with its longitudinal axis lying at the center line of a large Pyrex duct. The present paper reports the results obtained for a lean propane-air flame. The boundary velocity gradient, calculated at the stable flame positions by assuming a Blasius velocity profile in the boundary layer, is correlated with the propane-air ratio. The resulting curve is also compared with the curves obtained as flashback and blowoff limits for a Bunsen flame.

Theoretical Considerations

Consider the steady flow of a fluid along a slender rod of revolution. According to the boundary layer theory, a boundary layer starts to grow in thickness from the leading edge in the vicinity of the rod, where the viscous and the thermal effects are predominant. Outside of this region of developing boundary layer, these effects are negligible and the fluid can be treated as one without viscosity and thermal conductivity.

For a distance where the length Reynolds number is less than a certain critical value, which depends on the free stream turbulence level, thermal condition at the rod surface, and other factors, the boundary layer is laminar. Furthermore, if the boundary layer thickness is small compared with the rod radius, the velocity profile inside the boundary layer over the rod surface can be approximated by that inside the boundary layer over a flat plate. The latter problem has been studied by Blasius (3) and its solution can be found in various places in the literature, for example, (4).

In a first attempt to evaluate the present method of studying the flame stabilization in a boundary layer over a slender rod, the effects due to the presence of a flame on the velocity and temperature profiles inside the boundary layer are neglected. Because of this assumption, it seems that the uncertainty in the calculation of the boundary velocity gradient at the rod surface would not be greatly impaired if the Blasius solution is used. Hence, the boundary velocity gradient is calculated by the use of the expression

$$\left(\frac{\partial u}{\partial y}\right)_w = 0.332\left(\frac{u_\infty}{x}\right)\left(\frac{u_\infty x}{\nu_\infty}\right)^{1/2} \quad \ldots\ldots\ldots\ldots [1]$$

where u_∞ is the free stream velocity outside the boundary layer, ν_∞ is the kinematic viscosity of the fluid at free stream conditions, and x is the distance from the leading edge of the rod. Since the major component of the fluid in this investigation is air, the viscosity of the fluid is approximated by that of air in the calculation of the length Reynolds number in Equation [1].

Experimental Apparatus

Fig. 1 shows the schematic layout of the experimental apparatus. Air from a carbon-ring oil-free compressor and

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[4] Numbers in parentheses indicate References at end of paper.

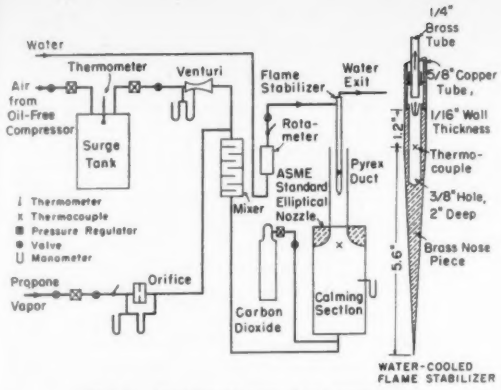

Fig. 1 Schematic layout of apparatus

propane vapor from a 100-lb tank are mixed in a mixer over a series of baffles. The homogeneous mixture then flows steadily through a calming section and an ASME standard elliptical nozzle into a Pyrex duct, in which is placed a flame stabilizer. The exhaust gases are expelled through the roof to the outside atmosphere by means of a hood and a chimney above the Pyrex duct.

The calming section, 10 in. ID and 21 in. long, consists of a disk-doughnut type of baffle, two blowout patches, three 200-mesh screens, a pressure tap, and a shielded thermocouple. The disk-doughnut baffle further insures a thorough mixing of propane and air. The blowout patches, consisting of 2- × 3-in. rectangular openings covered by household wax paper, are provided for safety reasons. A carbon dioxide line is also led into the calming section, so that carbon dioxide can be injected to control or extinguish the combustion.

The elliptical nozzle, reducing the flow cross-sectional area from a diameter of 10 in. to one of 3.16 in. (the inside diameter of the Pyrex duct), is made of plaster of Paris and reinforced with heavy wire. As this material cannot withstand intense heat, a 100-mesh screen is placed at the exit plane of the nozzle to prevent unstable flames from flashing back into the calming section. Tests are now being planned with this screen removed so as to insure a lower turbulence level in the flow.

Two Pyrex ducts, 24 and 46.5 in. long, are used in the present investigation to study the effects of duct length and of boundary layer growth next to the duct wall. They are both 3.16 in. ID.

Fig. 1 also shows the construction of the water-cooled flame stabilizer. It consists of $^5/_8$-in. thin-walled copper tubing 48 in. long, a brass nose piece 7 in. long, and $^1/_4$-in. brass tubing to carry away the cooling water. The nose piece is a cone-cylinder combination, with a $^3/_8$-in. hole, 2 in. deep. To measure the temperature of the cooling water, a chromel-alumel thermocouple is placed in this hole, with its wires threaded through stainless steel hypodermic tubing which is in turn placed inside the $^1/_4$-in. brass tubing. The coolant flows downward through the annulus and then upward through the central tubing. By maintaining given values of the coolant temperature inside the hole of the nose piece and at the entrance to the annulus, it is expected that the surface temperature of the flame stabilizer would be kept reasonably under control. However, for future investigations, the surface temperature should be independently measured and controlled.

As shown in Fig. 1, the air flow is measured by means of a venturi, and the propane flow by means of a square-edge orifice. The experimental error in these flow measurements is approximately ±0.5 per cent.

Both the position of the leading edge of the flame stabilizer and the position of the anchoring point of the stable, inverted-cone-shaped flame are measured by means of a vertical scale placed beside the Pyrex duct. The experimental error here is about ±0.1 in.

Discussion of Results

Effects of Stabilizer Position and Duct Length

As the stable flame position is determined by the profiles of the burning velocity and of the velocity of the unburned mixture inside the boundary layer over the slender rod, which acts as a flame stabilizer, it must be independent of the stabilizer position inside the duct, except for the extraneous effects due to the presence of another boundary layer which grows along the inside wall of the duct and due to dilution and flow divergence near the duct exit. The first few series of runs were thus made to investigate these effects. Fig. 2 shows the results of a typical series of runs, using ducts of two different lengths (24 and 46.5 in.). The fraction of the stoichiometric propane-air ratio, the free stream temperature, pressure, and velocity, and the stabilizer-coolant temperatures at the two ends of the flame stabilizer were held substantially constant, with values given in Fig. 2. Within a scatter of ±0.5 in., it is observed that the position of the flame from the leading edge of the rod is independent of the rod position inside the duct and of the duct length, unless the flame is vibrating or it is stabilized within 2 in. upstream of the duct exit. This indicates that, except under the latter circumstances, the stable flame position is determined by the flow characteristics in the boundary layer over the rod.

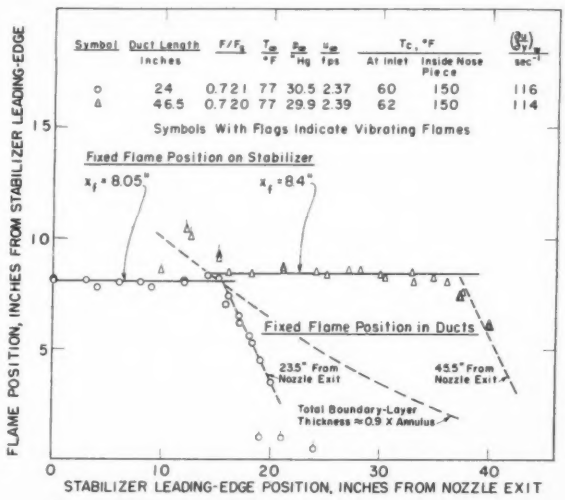

Fig. 2 Effects of stabilizer position and duct length

When a flame is stabilized within 2 in. of the duct exit, its shape indicates the presence of great divergence of flow of the unburned mixture. This flow divergence reduces the velocity of the unburned mixture near the stabilizer surface and thus allows the flame to stabilize there. Should this flow divergence be absent, the flame would be stabilized further downstream.

In Fig. 2 a dashed curve is also shown which indicates roughly the locus of the positions where the total thickness of the two boundary layers, one growing along the inside wall of the duct and the other along the slender rod, becomes equal to 90 per cent of the annulus. (The boundary layer thickness is defined as the distance normal to the wall where the velocity equals 99 per cent of the free-stream value.) Again, the thickness is calculated by assuming that the Blasius solution can be used to estimate the boundary layer growth along the rod and the duct. This particular dashed curve indicates that, at the point of stable flame attachment to the rod, the total boundary layer thickness varies on either side of the 90 per cent of the annulus. Although the estimate for the boundary layer growth is rather crude because of the large thickness compared with the annulus, it can still be concluded that the

effect on the flame position, due to the boundary layer along the duct, is negligible.

The nature of the vibrating flame was not studied in the present investigation.

Effect of Stabilizer-Coolant Temperature

As the burning-velocity profile in the vicinity of the rod surface is influenced by the surface temperature, it is expected that the boundary velocity gradient at the stable flame position depends also on the surface temperature. Since the quenching distance decreases with increasing surface temperature, it is also expected that the boundary velocity gradient increases with the surface temperature. This is illustrated in Fig. 3 for two specific sets of values of the propane-air ratio and the free stream velocity. It should be noted, however, that the abscissa there represents the temperature of the cooling water inside the hole of the nose piece of the rod and not the actual surface temperature. Since the latter temperature is not measured in the present investigation, the former temperature is used in the plot on the assumption that they are simply related when the inlet temperature of the coolant near the trailing edge of the rod is held constant. As this assumption is not a good one, it is planned to control and measure the surface temperature of the rod in the future.

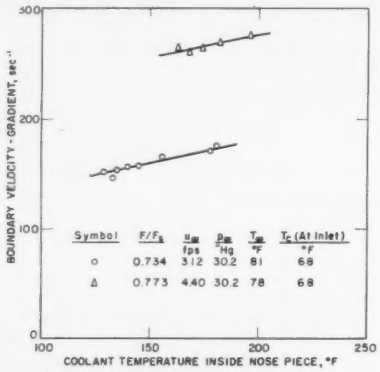
BOUNDARY VELOCITY - GRADIENT, sec⁻¹
COOLANT TEMPERATURE INSIDE NOSE PIECE, °F
Symbol F/Fs u∞ fps p∞ "Hg T∞ °F Tc (At Inlet) °F
o 0.734 312 30.2 81 68
Δ 0.773 440 30.2 78 68

Fig. 3 Effect of stabilizer-coolant temperature

It is to be noted that the values of the boundary velocity gradient, given by the curves of Fig. 3, at a coolant temperature of 150 F inside the nose piece, are about 25 per cent smaller than those given later in Fig. 5 for the corresponding propane-air ratios. However, the variation of boundary velocity gradient rather than its absolute value is the point of interest here, since the absolute values in these preliminary experiments may have been affected by the technique of manometer operation or by a variation in propane quality. (The later quantitative tests used a different bottle of propane.)

Effect of Free Stream Velocity

As the stable flame position is determined by the burning-velocity profile and the flow characteristics in the boundary layer over the rod, the effect of free stream velocity on the stable flame position should be completely expressed by Equation [1], with the value of the boundary velocity gradient remaining constant for a given burning-velocity profile. Fig. 4 shows a plot of the boundary velocity gradient versus the free stream velocity. Although a slight increase of the boundary velocity gradient with increasing free stream velocity is indicated there, it seems difficult to justify such a conclusion, because of the small range of variation of the free stream velocity (being limited by the short Pyrex duct) and the large effect due to even a small variation of the propane-air ratio. In other words, within the experimental accuracy of ±10 per cent observed later in Fig. 5, it may be concluded that, for a given burning-velocity profile, the boundary velocity gradient is independent of the free stream velocity.

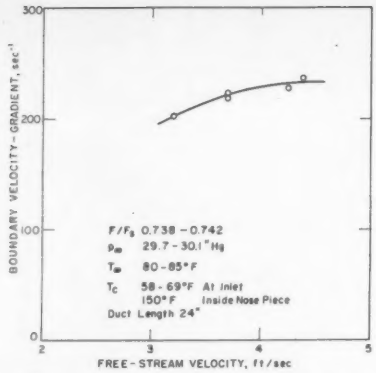
BOUNDARY VELOCITY - GRADIENT, sec⁻¹
FREE-STREAM VELOCITY, ft/sec
F/Fs 0.738 - 0.742
p∞ 29.7 - 30.1 "Hg
T∞ 80 - 85°F
Tc 58 - 69°F At Inlet
150°F Inside Nose Piece
Duct Length 24"

Fig. 4 Effect of free stream velocity

Effect of Propane-Air Ratio

Fig. 5 shows the variation of the boundary velocity gradient versus the fraction of the stoichiometric propane-air ratio. The velocity gradient is calculated for a given propane-air ratio on the basis of the mean stable flame position over a range of different rod positions and generally for more than one free stream velocity. Although the temperatures of the coolant for the flame stabilizer at the inlet and inside the nose piece were held substantially constant, there is no assurance that the surface temperature of the stabilizer would be constant. This probably accounts partially for a scatter of ±10 per cent in Fig. 5.

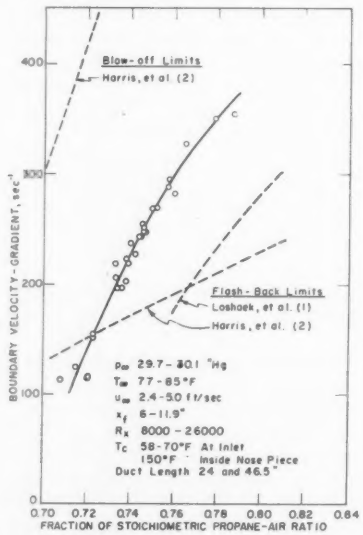
BOUNDARY VELOCITY - GRADIENT, sec⁻¹
FRACTION OF STOICHIOMETRIC PROPANE-AIR RATIO
Blow-off Limits
Harris, et al. (2)
Flash-Back Limits
Loshaek, et al. (1)
Harris, et al. (2)
p∞ 29.7 - 30.1 "Hg
T∞ 77 - 85°F
u∞ 2.4 - 5.0 ft/sec
xf 6 - 11.9"
Rx 8000 - 26000
Tc 58-70°F At Inlet
150°F Inside Nose Piece
Duct Length 24 and 46.5"

Fig. 5 Effect of propane-air ratio

Again, no consistent trend of the variation of the boundary velocity gradient at a given propane-air ratio versus the free stream velocity is observed.

Also shown for comparison in Fig. 5 are the curves of Loshaek et al. (1) for flashback limits and those of Harris et al. (2) for both flashback and blowoff limits. These curves were obtained by studying the stability of flames near the exit of a Bunsen burner. The comparison indicates that the boundary velocity gradients for stable flames determined in the present investigation have reasonable values.

It is considered that the quantitative conclusions from this type of experiment have more validity as a measure of boundary layer flame stabilization than conclusions drawn from two curves lying so far apart as the flashback and blowoff curves of a rim-stabilized Bunsen flame.

(*Continued on page 48*)

Stabilization of Premixed Propane-Air Flames in Recessed Ducts

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An alternate method of flame stabilization is proposed in an effort to obviate the pressure drop associated with bluff body flameholders. The proposed method consists of using recesses in the walls of the test chamber. It is shown that use of such recesses results in a wider range of stabilization limits and higher blowout velocities as compared to an arbitrary V-gutter flameholder, while indicating a pronounced decrease in the associated pressure drop.

Introduction

THE problem of flame stabilization in gas streams is of considerable concern both theoretically and practically. In general it is customary to stabilize a flame by means of a bluff body, commonly called a flameholder. In this paper, the authors describe experimental results obtained when premixed propane-air flames were stabilized by virtue of recesses incorporated into the test section. Technologically, a flame such as this which occurs adjacent to the walls may at the present be somewhat undesirable because of the drastic heating effects which would prove to be quite detrimental in an engine. However, the technique of using wall flameholders does offer certain possibilities in practice as well as in laboratory studies.

When a wall flameholder is used to stabilize a flame in a combustible gas stream, separation of the boundary layer occurs near the leading edge of the recessed contour in the wall, creating a pocket of recirculating gases. Hot gases in the recess ignite the adjacent fresh mixture in the flow stream and initiate a flame front which travels across the chamber. The recess is partially or completely filled with flame, indicating a high rate of influx of combustible mixture into the recess volume. The recess is completely filled for rich mixtures of fuel and air; but as the mixture is made lean a dark zone appears and increases in size. The dark zone is believed to consist of completely burned gases.

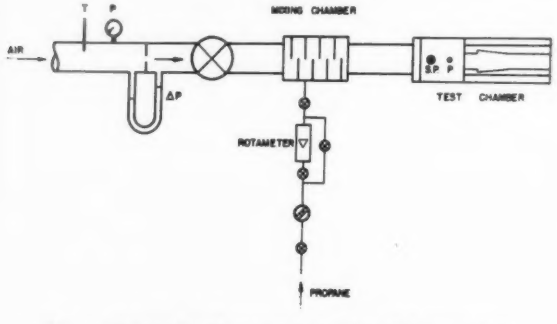

Fig. 1 Schematic diagram of experimental apparatus

The apparatus which was used is shown schematically in Fig. 1. It is hoped that this figure is self-explanatory and hence will not be further discussed. Fig. 2 is an atlas of the various recesses or flameholder profiles which were studied in the investigation. One each of flameholders 1 though 12 were placed in the lower and upper walls of the test chamber, thus forming recesses in its walls. It would have been possible to study the various flameholders by using only one of each fixed at either the top or bottom of the test section, since preliminary work showed that the blowout performance is the same regardless of whether one or two flameholders is used. However, a pair of each flameholder was installed in order to ascertain adequate burning of the air and fuel mixture within the length of combustion chamber used. The V-gutter included in Fig. 2 was used for comparative purposes, as will be explained later. Commercial propane was used and the mixture was not preheated but was at a temperature of approximately 70 F for all tests. The inside cross-sectional dimensions of the chamber are 1 in. × 1 3/4 in. with the flameholders in place. A change from a 2-in. circular section to the chamber cross-sectional dimensions occurs 9 in. upstream from the recess. The recess or the step in the flameholder is located 9 in. from the open end of the chamber. No attempt was made to control or measure the mixture turbulence in the chamber or the temperature of the metal flameholders.

Experimental Data

It was found that practically all of the flameholders would stabilize a flame, some with a wider stability range than others, however. Some of the data are grouped in Fig. 3 in which the mixture velocity entering the test section at blowout is plotted against the equivalence ratio. The velocities shown are subject to an estimated average error of 3 per cent and the equivalence ratios to an average error of 5 per cent. The blowout curves are incomplete at the top because of an insufficient supply of air during the time of the investigation. As may be noted from Fig. 3, the performance of flameholders 1, 2, 3, 4, 5, 7, and 8 may be represented by the same curve, although they have somewhat different geometric configurations. This is quite interesting because the test section is relatively small and thus differences in flameholder geometry and fuel characteristics are more prone to be noticeable. From Fig. 2 it can be seen that flameholder 7 differs from flameholder 2 only in that it has a more gradual slope at the downstream end of the recess. The similarity in their performances, as evident from Fig. 3, indicates that the degree of the slope of the downstream end of the recess does not affect flame stabilization appreciably. It can also be noted that the radius at the top of the upstream end of the recess of flameholder 8 has little effect on performance as this is the only difference from flameholder 1. During the experiments it was noticed that flameholders 4 and 7 operated somewhat roughly at higher velocities.

Fig. 4 shows a comparison between the performance of flameholders 1 and 6. As can be seen, flameholder 1 has wider stability limits than 6, which manifested rough operation on both lean and rich mixtures. A possible explanation

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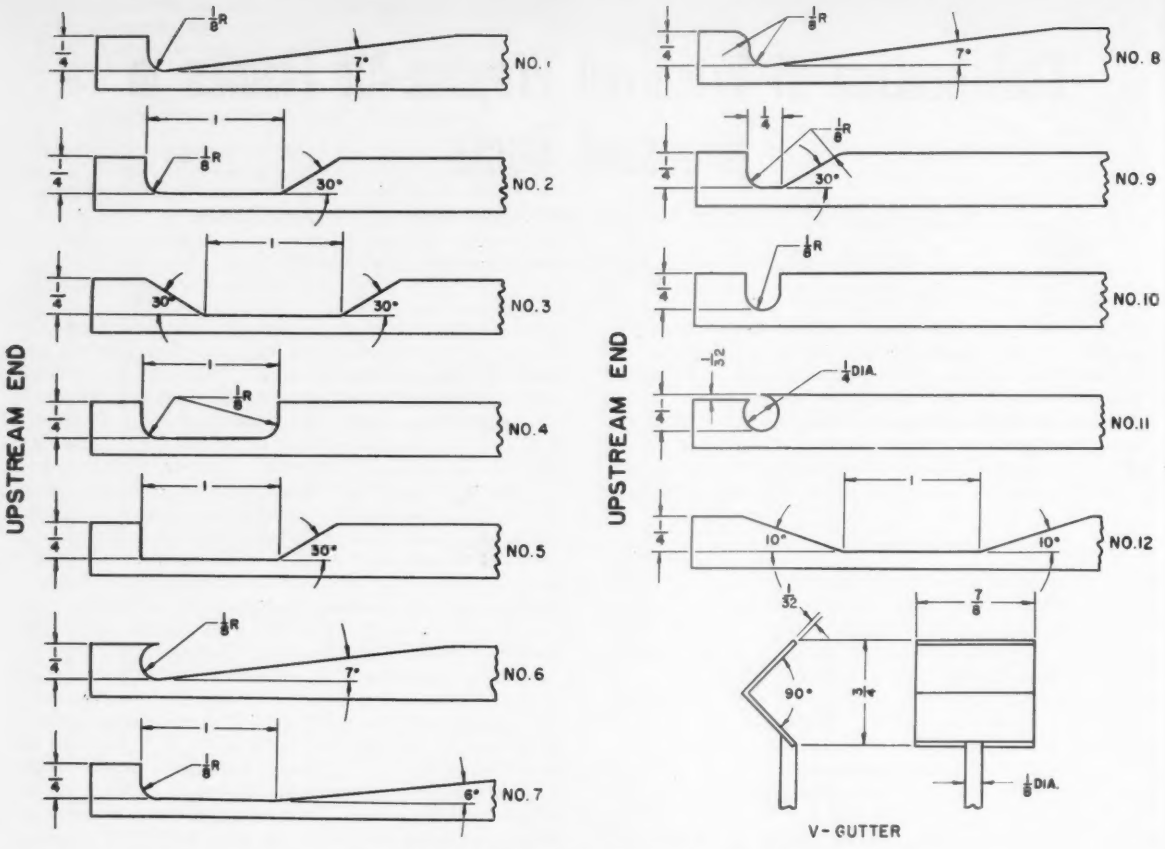


Fig. 2 Atlas of flameholder contours investigated

of the rough operation might be that vortices are created by the semicircular end of the recess thus causing a disturbance of the recirculating hot gases. It may be surmised that as a result the transport of heat and of chain carriers does not occur in a uniform manner.

Fig. 5 shows a comparison between the performance of flameholders 2 and 9. The only difference between the two is that flameholder 9 has a shorter recess length. The latter operated very roughly and pulsed appreciably and did not perform as well as 2. This indicates the necessity of having sufficient recirculation volume for achieving good stability.

Flameholders 10, 11, and 12 did not stabilize a flame. In the case of flameholders 10 and 11, this is probably due to a lack of sufficient recirculation volume. In the case of flameholder 12, the slope of the recess on the upstream side probably was not great enough to cause separation of the flow as it passed the recess. Therefore, the mixture diverged and filled the recess, thereby preventing any backflow or recirculation.

Direct and schlieren photographs of the flame anchored by flameholder 3 are shown in Figs. 6 and 7. The schlieren pictures were taken at an average time of 6 microsec by means of a high voltage flash. The photographs shown for this flameholder are typical of what occurred with the others. It can be seen that the flame originates from the point where the step in the wall begins. The flames from the upper and lower holders then spread across the chamber as the mixture flows downstream, and eventually the two flame fronts meet. The rate of flame spreading is less than that for a V-gutter; this apparently is due to the lack of turbulence which is created by such flow obstructions. Figs. 6 (a) and 7 (a) are for a mixture with an equivalence ratio of approximately 0.8 and a velocity of 110 fps, and Figs. 6 (b) and 7 (b) are for a

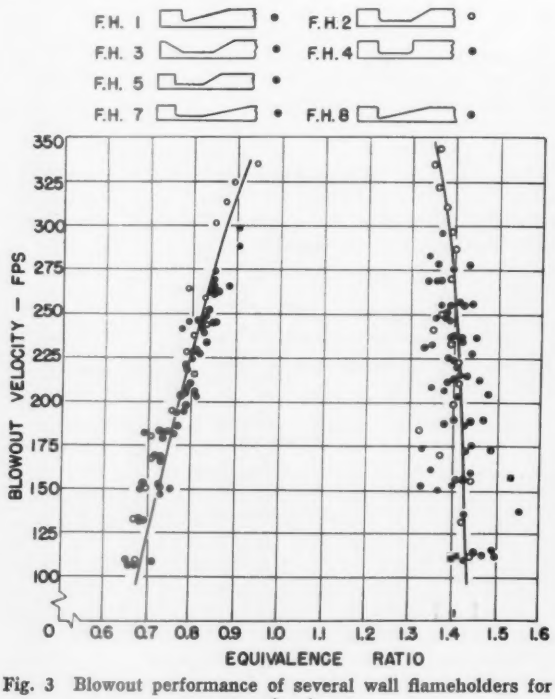


Fig. 3 Blowout performance of several wall flameholders for propane-air mixtures

mixture of equivalence ratio of approximately 1.3 and the same velocity. The flame fronts meet sooner as the mixture

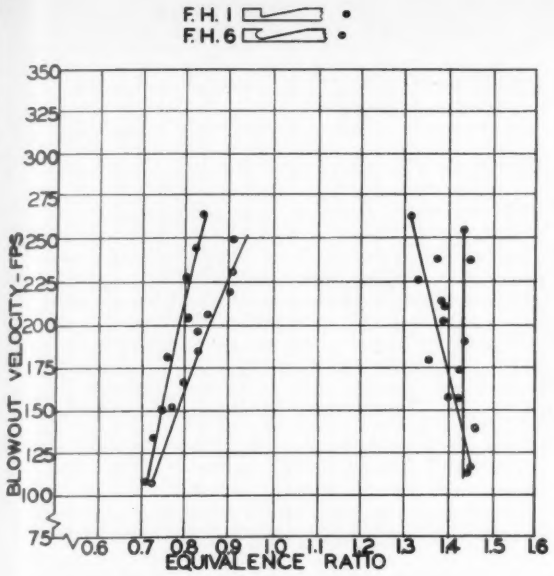


Fig. 4 Blowout performance curves for flameholders 1 and 6 for propane-air mixtures

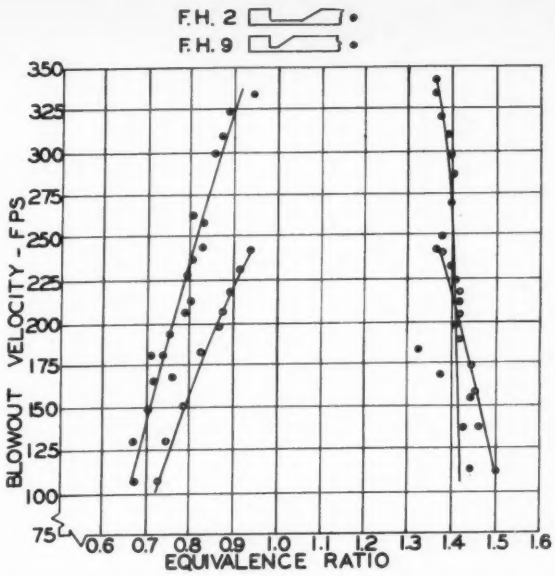
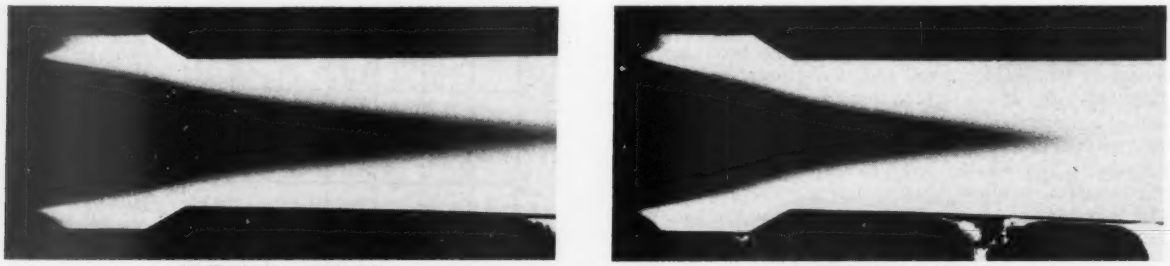


Fig. 5 Blowout performance curves for flameholders 2 and 9 for propane-air mixtures

becomes richer. The schlieren pictures show that the spreading of the flame into the unburned mixture is not as orderly a process as the direct flame pictures seem to indicate. Similar flame front distortions were observed by Zukoski (1)[4] in lean and rich flames stabilized on circular cylinders. One possible cause of the uneven flame front could be irregular flow of the approaching mixture. Another possible cause of the distorted flame front near the flameholder is occurrence of transverse oscillations. When testing the various flameholders, a recirculation zone could be noticed visually in the upstream end of the recess. This was especially evident when lean mixtures were being burned.

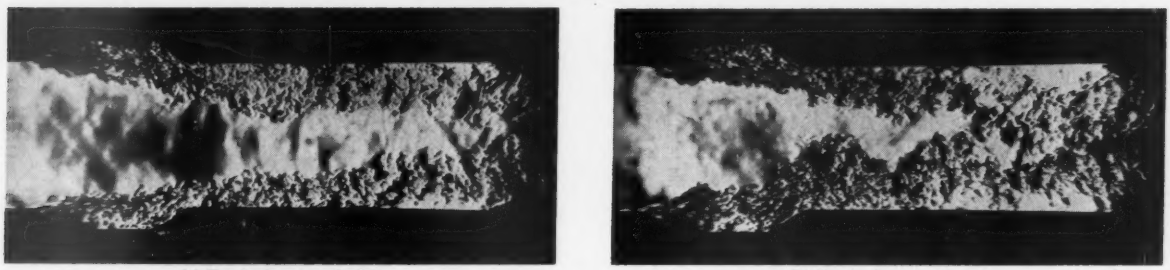
The effect of depth of the recess on the performance of wall flameholders is shown in Fig. 8. Flameholder 2 was used for this series of tests, and the depth of the recess was varied by milling away stock from the top surface. The flameholder was then shimmed up from the bottom to keep the chamber cross-sectional area constant. This flameholder was first tested at a recess depth of $^1/_4$ in. with the results mentioned earlier in the discussion. At $^3/_{16}$-in. depth, the stability limits were reduced considerably from those obtained at $^1/_4$-in. depth, although the maximum velocity was still not obtained, due to an insufficient supply of air. When the depth was reduced to $^1/_8$ in., the highest velocity at which a flame could be stabilized was 195 fps. At 33 sec in. depth, the maximum velocity was reduced to 160 fps. It is evident, therefore, that the depth of the recess and thus the volume of recirculation are significant variables affecting stabiliza-

[4] Numbers in parentheses indicate References at end of paper.



(a) Equivalence ratio 0.8 (b) Equivalence ratio 1.3

Fig. 6 Direct photograph of propane-air flame stabilized on flameholder 3 for approach velocity of 110 fps; flow from left to right



(a) Equivalence ratio 0.8 (b) Equivalence ratio 1.3

Fig. 7 Schlieren photographs of propane-air flame stabilized on flameholder 3 for approach velocity of 110 fps

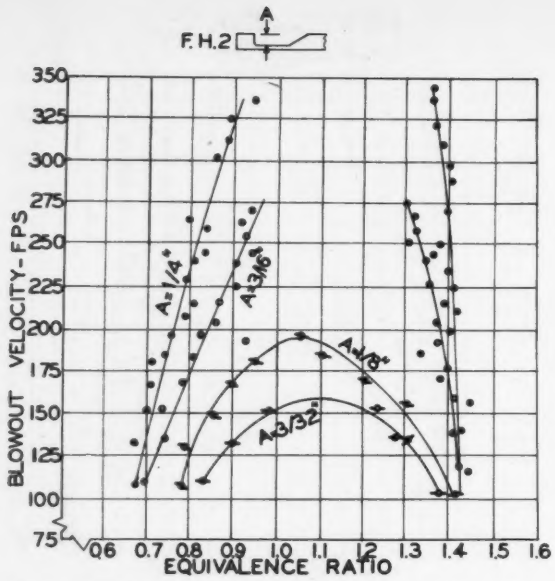


Fig. 8 Effect of recess depth of flameholder 2 on blowout velocity for propane-air mixtures

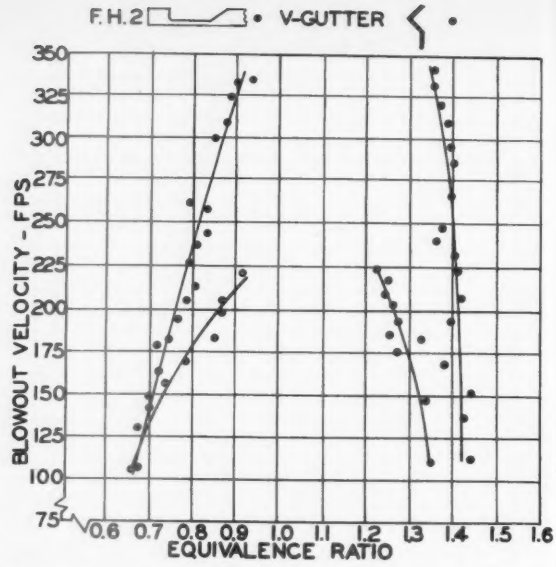


Fig. 9 Comparison of blowout performance for 90 deg V-gutter having 37 1/2 per cent blockage with wall flameholder 2 for propane-air mixtures

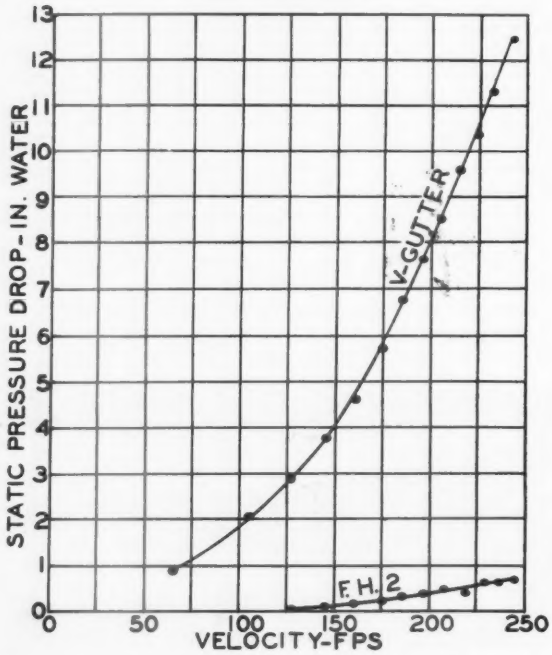


Fig. 10 Comparison of static pressure drop in absence of combustion for 90 deg V-gutter with 37 1/2 per cent blockage and wall flameholder 2

tion in this series of tests. If the depth of the recess in the wall flameholder is thought of as being analogous to the size of bluff body flameholders, these results could have been expected since it has been shown by Longwell et al. (2) and DeZubay (3) that increasing the size of bluff body flameholders increases their performance.

The influence of heat transfer on the performance of the wall flameholder should be considered. It is known that the temperature of the approach mixture and of the recirculation zone affect the performance of any flameholder. Therefore, variations in the stability limits with modifications of flameholder geometry may be caused by different rates of heat transfer and accompanying changes in gas temperature. This may, in part, explain the variation of performance with reductions in flameholder depth. The superior performance of flameholder 1 compared to flameholder 6 may be due to greater heat loss because of the lip of the latter configuration.

As a comparative test, the 90-deg V-gutter shown in Fig. 2 was studied. The performance and the pressure drop curves for flameholder 2 and such a V-gutter are shown in Figs. 9 and 10. It can be seen that the wall flameholder performed considerably better than the V-gutter tested. It not only resulted in wider stability limits, but also operated much smoother. The V-gutter had a 37 1/2 per cent blockage area but otherwise was not designed for optimum performance. The V-gutter was 7/8 in. wide and centered in the 1-in.-wide chamber and thus exhibited lower performance than a truly two-dimensional V-gutter of the same dimensions. Further, the position of the supporting strut may have had a detrimental effect on the performance of the V-gutter. The flameholder pressure drop curves, Fig. 10, show the pressure drop due to flow past the flameholder plotted against gas velocity. These data were taken without combustion, which would simulate the condition in a jet engine afterburner when the afterburner was not in use. Air alone was used in obtaining the data. Even though the blowout performance of the V-gutter was below that of the wall flameholder, it can be seen from the curves that there is associated with it a much greater pressure drop than with the wall flameholder. At 225 fps, the pressure drop due to the wall flameholder is approximately 5 per cent of that caused by the V-gutter. The V-gutter acts as a restriction to the flow, causing this large pressure drop. There is no physical restriction with wall flameholders. This indicates the advantages which could be gained if wall recesses were to be adopted for use in jet engine afterburners in place of V-gutters, because the usual loss of thrust when the afterburner is not in use would be almost negligible.

Conclusions

Technologically, a flame stabilized by virtue of a recessed wall may be undesirable because of the drastic wall heating which would result and thereby be detrimental to the engine. However, the reduction of pressure drop makes this scheme attractive fluid dynamically. Another limitation of wall-

(*Continued on page 43*)

Dynamics of a Variable Thrust, Pump Fed, Bipropellant, Liquid Rocket Engine System

MARVIN R. GORE[1] and JOHN J. CARROLL[2]

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The principal dynamic characteristics of a pump fed, variable thrust, bipropellant, liquid rocket engine system are established over a broad thrust range by analog computer simulation of the nonlinear physical elements. Several basic types of control elements are studied, and a control system capable of meeting a stringent set of transient performance requirements is synthesized. The results of this study are of general applicability to rocket engine system design since the effect of the intrinsic system nonlinearities upon performance at various thrust levels is established. Transient and frequency response characteristics of the nonlinear system are determined for both small and large changes in thrust level. Correlation between the general nonlinear equations and the linear equations which describe the rocket engine system for perturbations about any fixed thrust level is shown.

Nomenclature

A = area
C^* = thrust chamber characteristic velocity
G_c = controller transfer function
g = gravitational constant
H = head rise
I = turbine-pump assembly inertia
K = gain of element
L^* = thrust chamber characteristic length
P = pressure, psi
Q = propellant volumetric flow
R = propellant line resistance
r = turbine wheel radius
s = complex operator
T = time constant, sec
t = initial or real time
U = turbine wheel peripheral speed
V = gas spouting velocity
W = propellant weight flow
X = actuator position
ϵ = error = $P_{C\,REF} - P_C$
η = efficiency
θ = turbine speed
μ = torque
τ = combustion lag time

Subscripts

a = actuator
b = base frequency or time
c = chamber
C = controller
D = discharge
f = fuel
g = gas source
N = turbine nozzle
OL = open loop
ox = oxidizer
P = pump
s = suction
T = turbine
t = thrust chamber throat
V = throttle valve
v = velocity
p = proportional

Presented at the ARS Semi-Annual Meeting, Cleveland, Ohio, June 17–21, 1956.

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[2] Design Engineer, Liquid Engine Division. Mem. ARS.

[3] Numbers in parentheses indicate References at end of paper.

Introduction

THE development of an integrated power control system for a bipropellant, liquid rocket engine, designed to meet a stringent set of transient and steady state performance requirements, can be accomplished effectively only if the principal dynamic characteristics of the rocket engine are fully understood. Up to the present these characteristics, from a control system viewpoint, have not been clearly delineated. On the one hand, linear differential equations, based upon perturbation methods, have been developed; but, the region of validity for these equations has not been established. On the other hand, the potential complexities, both in mechanization and in interpretation, of analog computer simulation of all of the rocket engine and control system parameters have led to the espousal of the trial and error method of design.

A more rational course lies somewhere between these extremes. In this paper, by utilization of a set of nonlinear differential equations and graphical relationships which design and test experience have demonstrated to be valid for power control system synthesis, the principal dynamic characteristics of a variable thrust, pump fed, bipropellant, liquid rocket engine are established, and the performance of the rocket engine with typical controllers is analyzed. The specific objectives of this study are:

1 To outline the role of dynamic analysis in integrated rocket engine control system design.

2 To establish the principal dynamic characteristics of a liquid rocket engine.

3 To demonstrate the performance of a liquid rocket engine control system with several basic types of controller.

4 To extend the validity of the linear perturbation equations by correlating them with analog computer simulation of the general nonlinear system equations (1).[3]

Integrated System Design and Dynamic Analysis

In keeping with the first of the above objectives it is necessary to define the integrated system design concept as related to rocket engine power control systems. This will establish the role of dynamic analysis in rocket engine system development. A rocket engine control system, like other control systems, is composed of a group of components which, when properly combined, perform as a unit for a definitive purpose. Such a system may be rendered ineffective by arbitrary overemphasis of the importance of a single component or design concept. Each element should be selected, or designed, not only on the basis of its suitability for performing a specific function, but also on the basis of its integrability with the other elements.

The design of components to meet specified performance requirements is subject to basic physical constraints. Certain components are relatively unalterable because their de-

sign is, to a large extent, dictated by environmental and power requirements. However, frequently some parameters of even the relatively unalterable elements are flexible and can be varied over a range. The final fixing of these parameters should be based upon an appraisal of their effect on the over-all system performance.

The major components of a variable thrust, liquid, bipropellant, pump fed rocket engine, as shown in Fig. 1, are (a) the thrust chamber, (b) propellant pump, (c) propellant lines, (d) turbine, (e) flow control valves, and (f) thrust controller. The characteristics of the turbine, pumps, lines, and thrust chamber are, to a large extent, fixed because of the design constraints imposed by the thermodynamic and materials problems associated with the selection of propellants to meet weight and steady state thrust requirements More flexibility is possible in the selection of the type of controller and in the location of the control point. The controller must be designed to compensate for the tolerances inherent in the fixed components and to meet the transient performance requirements.

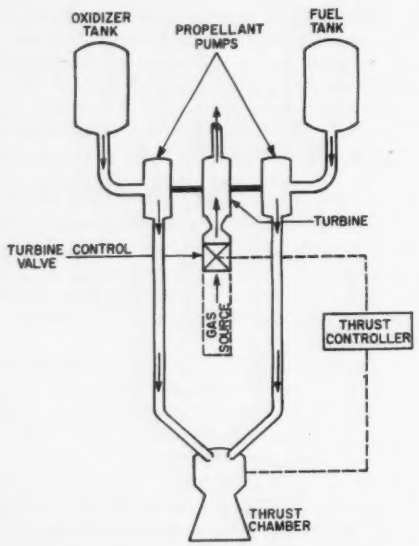

Fig. 1 Major components of a variable thrust, pump fed, bipropellant, liquid rocket engine system

The performance specifications for any system are generally of two types, static and dynamic. The static performance requirements are related to system absolute stability, i.e., to the selection of the unalterable components which will yield a balance between power developed and power required at specified operating levels. It is of no significance to speak of dynamic stability without possessing static stability.

Dynamic stability relates to the manner in which a system responds to changes in some reference input (e.g., the manner in which the thrust level of a rocket engine is varied from one static equilibrium point to another) and the ability of the system to compensate for unwanted disturbances (e.g., to maintain an operating level within a fixed percentage variation in spite of load changes or uncertainties introduced by manufacturing tolerances or minor malfunctions). Dynamic performance customarily is specified in terms of such criteria as (a) maximum response time, (b) maximum overshoot, (c) maximum steady state error.

These requirements are frequently severe since they may reflect the performance specifications of a larger unit, of which the rocket engine is a part, or may be imposed by the physical limitations of the rocket engine itself. For example, a very rapid response time, i.e., a capability for rapidly changing thrust level, may be specified on the basis of the performance required of the carrier of the rocket engine. Simultaneously, a limitation on overshoot may also be imposed in order to avoid the critical speed of the turbine wheel. Typical steady state error specifications, i.e., thrust accuracy after the initial transient has subsided, may well be of the order of a few per cent.

Obviously, these requirements, more often than not, are antithetical. Hence, the careful synthesis of a feedback control system is required in order to integrate and guide the over-all system design and to circumvent as many problems as possible before the rocket engine reaches the test stand.

In order to achieve an integrated system design it is necessary to establish a research and development program which, although following a logical sequence, is flexible in execution. Such a research program consists of four major phases:

1 Establishment of performance requirements and preliminary design specifications.
2 Preliminary design study.
3 Prototype component development and testing.
4 Prototype system testing.

Subsequent to the establishment of performance specifications, the flow of information is not unidirectional. Flexibility is achieved because of the feedback between stages. For example, the preliminary design study, while dictating the preliminary design specifications for all components, must be capable of revision, if necessary, as soon as test data on individual components becomes available. The effect of actual component performance in relation not only to the component itself, but also to over-all system performance must be evaluated and any necessary design changes made.

The role of dynamic analysis and simulation in rocket engine system development is now clear. Since it it not possible, before prototype system testing, to ascertain the exact performance of the complete system, it is desirable to establish a continuing system study which, with an increasingly greater degree of accuracy, can predict the actual performance of the prototype unit and which can minimize the necessity for costly trial and error experimentation by guiding the integrated system design.

The principal techniques for dynamic analysis, in order of increasing complexity, are (a) linear system analysis, (b) nonlinear system simulation, (c) direct simulation.

Linear system analysis does not necessarily require the use of an electronic computer. The set of system differential equations are linearized about one or more operating points, and the system dynamic performance is determined in the vicinity of these operating points.

The use of an electronic computer, such as an analog simulator, makes possible the inspection of system performance over a broad operating range because the nonlinear system equations can be simulated without recourse to linearization. Controller types can be simulated and control system synthesis can be executed with a rapidity impossible to achieve by hand calculation.

Direct simulation implies the incorporation of actual components in the analog simulator. In cases where this is not practicable the performance characteristics of the actual components may be determined by test and the simulator circuitry modified accordingly.

In rocket engine system development it is extremely desirable to extend the validity of the first two methods of dynamic analysis not only because tests are costly, but also because the required modifications may jeopardize the success of the entire development program if no indication of over-all system dynamic performance is obtained prior to the test phase.

The validity of the linear analysis is subject to two factors:

1 The accuracy of the particular system nonlinear equations in describing actual system performance.

2 The degree to which the nonlinear equations can be linearized (i.e., the magnitude of the region about an operating point for which the linear equations are valid).

If these two factors can be evaluated, then a preliminary

linear analysis can be very rewarding. It can establish the dominant system dynamic characteristics and can be of use in interpreting computer results and guiding a simulator program.

The validity of the nonlinear system simulation is dependent upon only the first of the above factors. The absolute degree of validity of these equations for any specific system can be determined only by actual test of the system components. However, if it is necessary to await this data before conducting a dynamic analysis, in rocket engine development, at least, it frequently is too late. Fortunately, experience with rocket engine component design and testing has over a period of years provided a source of information which, for purposes of power control system simulation, makes possible the establishment of a set of defining relationships which is sufficiently accurate for initial system synthesis and which can guide the integrated system development program successfully to a point at which the incorporation of the actual component characteristics, when established by tests, necessitates no major design modifications. When these more precise dynamic characteristics of the system elements are included in the simulator, additional applications or proposed modifications of the unit can be evaluated readily.

Rocket Engine Control System Dynamics

A basic power control system for a variable thrust, bipropellant, pump fed rocket engine is shown in Fig. 2. Thrust chamber pressure is sensed and compared with reference pressure which is proportional to the desired thrust level.

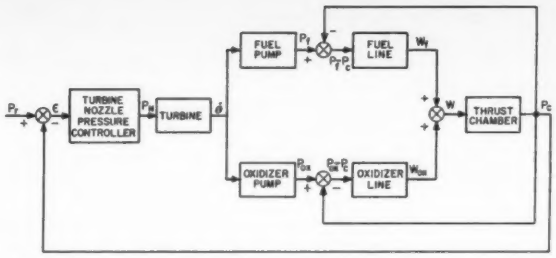

Fig. 2 Block diagram of a variable thrust, pump fed, bipropellant, liquid rocket engine system

The resultant error signal is supplied to a suitable controller which positions a control valve in the turbine gas supply duct in order to achieve correspondence between the actual and reference thrust levels.

The fundamental rocket engine dynamic equations are tabulated in Table I (2). These equations provide a representation of rocket engine performance over a large thrust range which is quite valid for a control system and transient performance study. The major assumptions implicit in these equations are:

1 Constant pump suction pressure, P_s.

2 Constant characteristic velocity for the turbine and thrust chamber gases, $c_T{}^*$ and c^*.

3 Turbine gas spouting velocity, V_T, not a critical function of shock or underexpansion.

Table 1 General rocket engine dynamic equations

Elements	Equation	Symbols
Propellant pump		
(a) Flow equation	$H = \dfrac{Pd - P_s}{\rho} = K_1\dot{\theta}^2 - K_2Q^2$	H = head rise P_D = pump discharge pressure P_s = pump suction pressure ρ = propellant weight density $\dot{\theta}$ = turbopump speed Q = propellant volumetric flow μ_P = pump torque required K = constant
(b) Power equation	$\dfrac{\mu_P}{\rho\dot{\theta}^2} = K_3 + K_4\dfrac{Q}{\dot{\theta}}$	
Turbine		
(a) Torque balance	$I\dot{\theta} = \mu_T - \mu_P$	μ_T = turbine torque developed I = turbine-pump assembly inertia W_T = weight flow of gas into turbine V_T = turbine gas spouting velocity η_T = turbine efficiency r = turbine wheel radius U = turbine wheel peripheral speed = $r\dot{\theta}$ A_N = turbine nozzle area P_N = turbine nozzle pressure $C_T{}^*$ = turbine gas characteristic velocity P_T = turbine exhaust pressure
(b) Turbine efficiency	$\eta_T = K_5\dfrac{U}{V_T}\left(1 - K_6\dfrac{U}{V_T}\right)$	
(c) Torque developed	$\mu_T = \dfrac{W_T V_T \eta_T r}{2gU}$ $= \dfrac{W_T}{g}(K_7 r V_T - K_8 r^2\dot{\theta})$	
(d) Weight flow of gas into turbine	$W_T = \dfrac{A_N P_N g}{C_T{}^*}$	
(e) Turbine gas spouting velocity	$V_T = K_9 C_T{}^*\left[K_{10} - K_{11}\dfrac{P_T}{P_N}\right]$	
Control valve		
(a) Weight flow of turbine gas	$W_T = K_{12} P_g A_V \psi$	P_g = source gas pressure A_V = throttle valve area ψ = compressible flow coefficient $\psi = (P_N/P_g)$ P_C = thrust chamber pressure G_C = controller transfer function
(b) Controller	$A_V = G_C(P_{C\,REF} - P_C)$	
Propellant lines	$P_d - P_C = RQ^2 + K_{14}\rho\dot{Q}$	R = total line flow resistance
Thrust chamber	$[\rho_f Q_f + \rho_{ox} Q_{ox}](t - \tau) = \dfrac{A_t g}{C^*} P_C + \dfrac{L^* A_t K_{15}}{(C^*)^2}\dot{P}_C$	L^* = thrust chamber characteristic length A_t = thrust chamber throat area C^* = thrust chamber characteristic velocity τ = combustion lag time

4 Combustion lag time τ independent of chamber pressure (3, 4).

5 Propellent lines sufficiently short for lumped parameter representation (5).

6 Constant turbine wheel blading coefficient.

7 Constant specific heat ratio for gases.

The interrelations between the various components and their defining equations are illustrated by the block diagram of Fig. 3.

The investigation of system dynamics consisted of two phases: determination of the dynamic properties of the basic rocket engine without a controller, and synthesis of a controller to meet the over-all system performance specifications.

During the first phase both "no flow" and "flow" conditions exist. That is, during the system starting transient, as the turbine accelerates there is no flow of propellants into the thrust chamber until the opening of the flow control valve is begun. The starting, or "no flow," transient is the most difficult phase of rocket engine operation to study because the system nonlinearities are much more severe during the starting transient than during changes of steady state thrust level. Effectively, the system damping and the system gain are radically altered during the transition from "no flow" to "flow."[4]

After the opening of the propellant flow control valve, the rocket engine is characterized by the set of equations of Fig. 3. Fig. 4 is the electronic analog mechanization of the equations of Fig. 3. The rocket engine control system, as shown in Figs. 2 and 4, is comprised of two major closed loops: (a) the intrinsic propellant line-thrust chamber loop, and (b) the external control loop. The characteristics of the propellant line-thrust chamber loop have been the subject of much investigation and have been discussed in numerous technical papers for a constant pressure source, e.g., pressurized propellant tanks (3, 4). Use of a turbine driven pump propellant supply does not alter the dynamics of this loop because of the low pass filter characteristics of the turbine, attributable to the turbopump assembly inertia and damping. However, it is necessary that this inner, line-thrust chamber loop be stable and that its effect upon the outer, or control, loop be considered. Hence, the following program was established for synthesizing a control system for a variable thrust, bipropellant, liquid rocket engine:

1 Determination of propellant lines–thrust chamber dynamic response characteristics.

2 Determination of dynamic response of the basic rocket engine.

3 Synthesis of a controller to meet the system performance requirements.

Propellant Lines–Thrust Chamber Loop Dynamic Characteristics

The rocket engine was assumed to have sufficiently short feed lines for a lumped parameter representation of the line-thrust chamber loop. The combustion lag time τ was assumed constant for any single system performance investigation. The study was repeated for a range of lag times compatible with experimental data obtained from tests of physical systems. The analog simulator diagram is shown in Fig. 5. A low frequency oscillator was used to obtain the open loop frequency response characteristics of the line-thrust chamber loop over a broad range of operating levels for both small and large perturbations about these operating levels. Fig. 6 is an open loop frequency response plot for the line-thrust chamber loop at the maximum and minimum operating levels. Both amplitude and phase are plotted. The amplitude curves are plotted in terms of relative amplitude ratio (i.e., the ampli-

[4] In this paper, only the behavior of a rocket engine at different flow levels and the nature of the system transient performance during change of flow level are discussed. However, the starting transient, including "no flow" conditions, has been studied in a more elaborate simulation and is the intended subject of a subsequent paper.

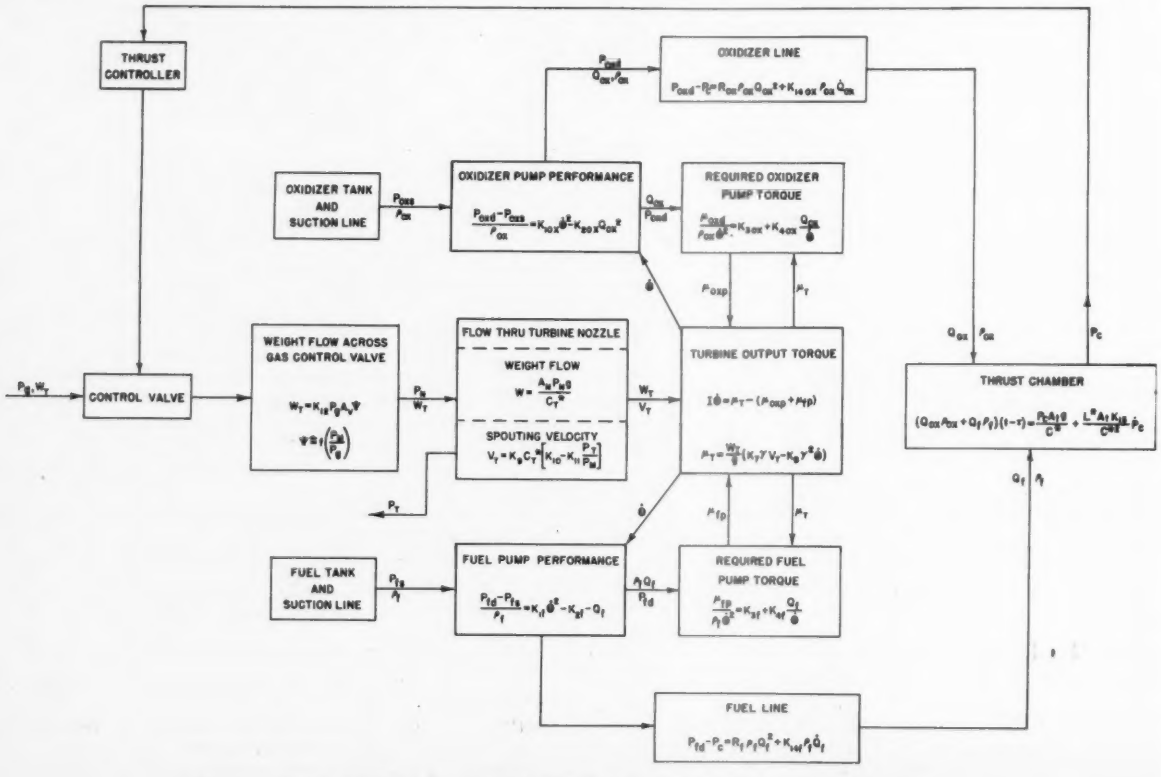

Fig. 3 System equations for a variable thrust, bipropellant, liquid rocket engine system

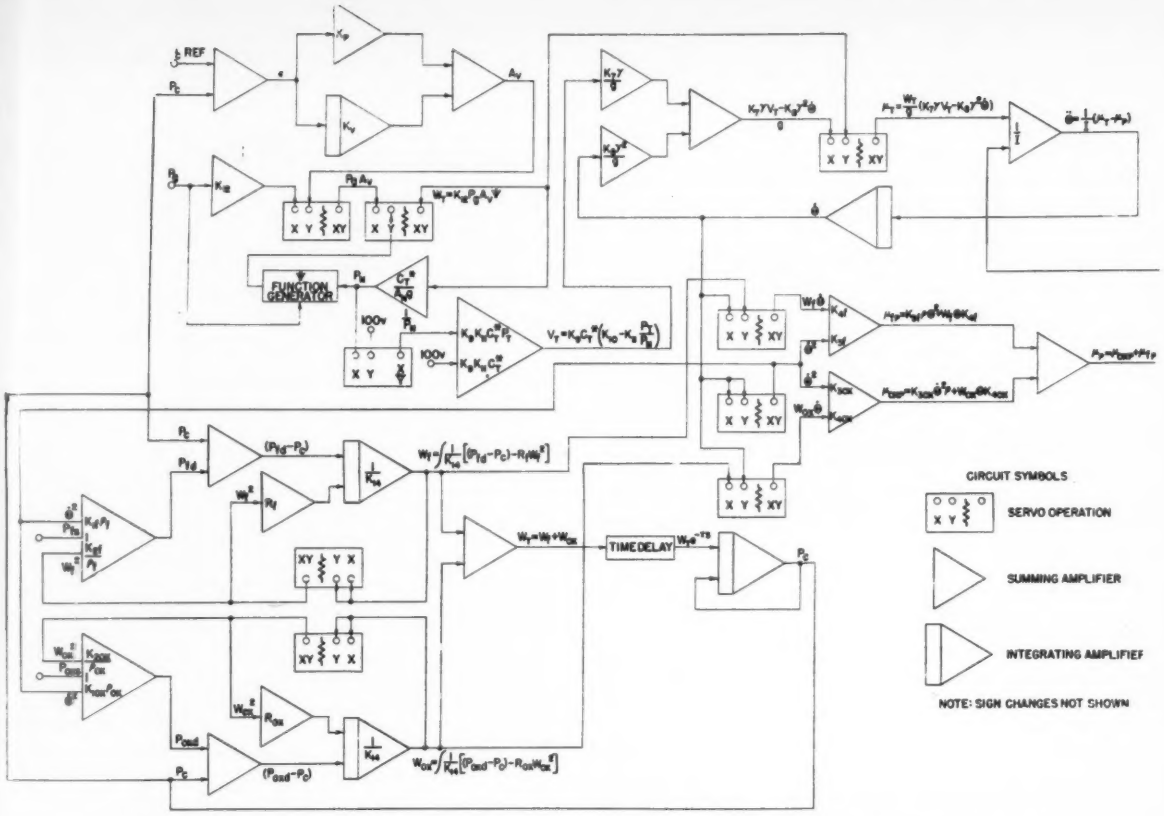

Fig. 4 Analog simulator diagram for a variable thrust, pump fed, bipropellant, rocket engine control system

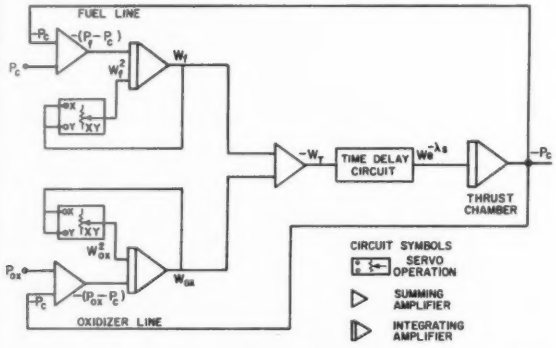

Fig. 5 Analog simulator diagram for investigation of propellant lines—thrust chamber dynamics

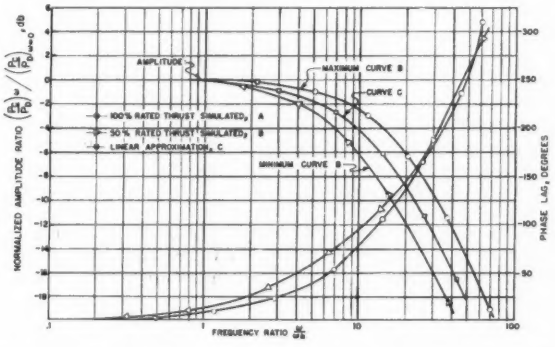

Fig. 6 Frequency response of line-thrust chamber loop, open loop

tudes at all frequencies are divided by the static, or zero frequency, amplitude). The frequency scale, for convenience, is similarly normalized by dividing by a base frequency. The same base frequency ω_b is used to normalize all frequency response curves.

Curve A is the normalized open loop frequency response obtained at the maximum thrust levels for an input P_D max $\pm\Delta P_D$; P_D max is the static value of pump discharge pressure corresponding to maximum rated thrust; ΔP_D is the sinusoidal perturbation induced about P_D max. No significant deviations from this curve were observed for ΔP as great as 33 per cent P_D max. Inspection of Curve A reveals that, although there is no significant amplitude attenuation up to a frequency ratio, FR, ω/ω_b of 10, there exists appreciable phase shift at much lower frequencies. This is compatible with the nonminimum phase properties characteristic of combustion lag time.

Curve B is the frequency response obtained at the minimum rated thrust level for oscillations of ± 10 per cent and ± 33 per cent about the minimum design pump discharge pressure P_D min.

The amplitude and phase characteristics of Curves A and B are similar over the frequency range of significance and an approximation, shown as Curve C, obtained by linearization of the line-thrust chamber equations at an intermediate thrust level is a valid equivalent for the normalized open loop frequency response thrust levels.

The normalized frequency responses shown in Fig. 6 are almost invarient with thrust level. However, as shown in Fig. 7, the step response of the line-thrust chambers loop becomes more oscillatory at the lower thrust levels. This decrease in stability must, then, be attributed to an effective open loop gain which increases as thrust level decreases. This increase in gain at the lower thrust levels conforms to the

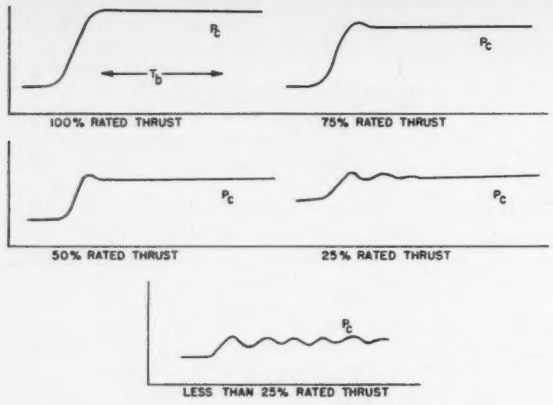

Fig. 7 Step response of line-thrust chamber loop at various thrust levels

mathematical expression for open loop gain derived by a linearized analysis of a similar system. The equation for the open loop gain K_{OL} is (Ref. 1)

$$K_{OL} = \frac{\bar{P}_c}{2\,(\bar{W}_f + \bar{W}_{oz})}\left[\frac{\bar{W}_f}{\bar{P}_f - \bar{P}_c} + \frac{\bar{W}_{oz}}{\bar{P}_{oz} - \bar{P}_c}\right] \quad \ldots\ldots [1]$$

where the barred quantities are equilibrium design values. The gains corresponding to the thrust levels of Fig. 7 are tabulated in Table 2. The closed loop gain $K_{OL}/(1 + K_{OL})$ is included.

Table 2 Line-thrust chamber dynamic response

Thrust level (maximum thrust, %)	Open loop gain	Closed loop gain	Step response for typical
100	0.80	0.44	very damped
75	1.03	0.51	damped
50	1.45	0.59	damped oscillation
25	2.60	0.72	oscillatory
<25	>2.60	>0.72	oscillations increase toward instability

A necessary but not sufficient condition for instability is that the open loop gain be greater than unity. The degree to which the gain may exceed unity before sustained oscillations occur is a function of the particular thrust chamber and line parameters. As shown in Fig. 7, severe oscillations occur below 50 per cent maximum rated thrust. This was established as the minimum design thrust level. The line-thrust chamber loop may be considered as a single element in the control loop. The gain of that element is the closed loop gain of the line-thrust chamber loop. Hence the gain of the control loop also exhibits a deviation, although less radical, with thrust level.

Basic Rocket Engine Dynamic Characteristics

The addition of the equations for the turbine, pumps, and gas flow control valve completes the analog of the basic rocket engine, i.e., of the unalterable components. If the gas flow control valve position Xa is varied sinusoidally and the corresponding changes in chamber pressure recorded, the open loop response of the basic rocket engine may be obtained. Fig. 8 is a series of such open loop response plots at the 100 and 50 per cent of full thrust operating levels and for both low and high values of combustion lag time. These plots illustrate the effects of the combined propellant line-thrust chamber and turbine pump dynamics. The phase shift at frequencies above $\omega/\omega_b = 5$ is due principally to the combustion lag time, and the attenuation of 6 db per octave, beginning at about $\omega/\omega_b = 1$, is due principally to the turbine-pump assembly time constant. The effect of combustion lag time is shown in Fig. 8 by the marked increase in phase lag for the higher value of τ. This phase shift imposes a limit upon the gain of a closed-loop control system because there must be sufficient attenuation at the frequencies at which phase lag is excessive to insure stability. The curves of Fig. 8 are, again, normalized; however, the change in the open loop gain of the complete rocket engine system with thrust level is apparent from the slopes of the curves of Fig. 9, in which equilibrium chamber pressure and turbine speed are plotted as functions of the gas flow control valve position. At the 50 per cent thrust level the gain is approximately twice that at the 100 per cent thrust level.

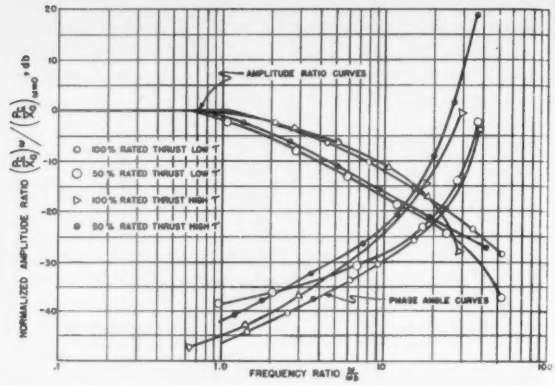

Fig. 8 Open loop frequency response of basic rocket engine

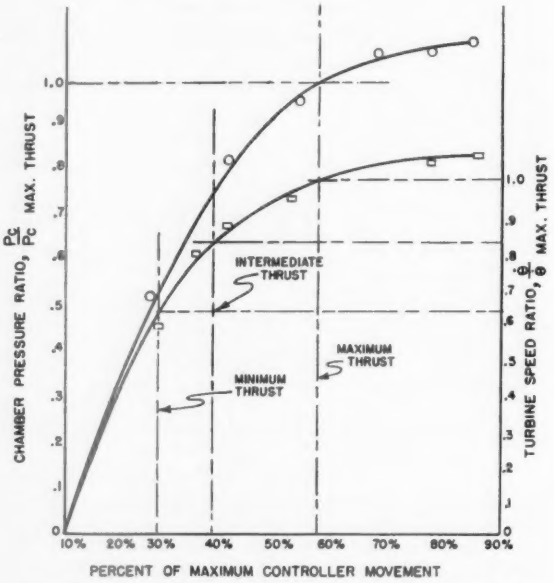

Fig. 9 Chamber pressure and turbine speed vs. control valve position

As a result of this phase of the over-all control system synthesis study it is possible to summarize the dynamic characteristics of the basic rocket engine:

1 Frequency responses for the nonlinear rocket engine analog for perturbations as large as 33 per cent of steady state value are similar to responses obtained from a set of linearized equations.

2 The frequency sensitive amplitude and phase characteristics of the basic rocket engine do not vary appreciably with thrust level.

3 The frequency insensitive characteristic, i.e., the gain, of the basic rocket engine increases as thrust level decreases.

4 The control system synthesis problem is most critical at the lower thrust levels because of the decrease in stability at those levels.

Rocket Engine Control System Synthesis

After determination of the dynamic characteristics of the basic rocket engine, the synthesis of a suitable closed loop power control system for the rocket engine was undertaken. For this purpose three basic types of controller were investigated. These were (a) velocity, (b) proportional, and (c) velocity plus proportional. A general relationship between controller output X_a and error ϵ is

$$X_a = K_p\epsilon + K_v\int \epsilon\, dt \quad \ldots\ldots\ldots\ldots\ldots\ldots [2]$$

where

K_p = proportionality constant
K_v = velocity constant

The transfer functions, i.e., input-output ratios, for velocity, proportional, and proportional plus velocity controllers are easily obtained from Equation [2] if negligible controller time delays are assumed. However, actual controllers are physically characterized by time delays. In Table 3, the transfer functions for three basic controller types are tabulated for zero and for nonzero controller time lags, and in Fig. 10 three hydraulic controllers of the type described above are drawn schematically. Except for limits on actuator motion, the controllers used to synthesize the control system were described by these transfer functions.

The system performance with the various types of controllers was evaluated by assuming as a performance specification (a) a maximum turbine speed overshoot of 5 per cent in going from 50 to 100 per cent rated thrust, and (b) a rise time not to exceed T_b, where $T_b = 1/\omega_b$. The relative effectiveness of each basic controller is summarized in Table 4 for controller gain adjusted to yield 5 per cent overshoot.

A proportional controller, while providing the optimum rise time, may be incapable of satisfying the steady state error requirements. This is due to the fact that changes in ambient operating conditions or changes in the system parameters due to minor malfunction or changes in operating level can only be calibrated out at one operating point and under one set of ambient conditions.

A velocity controller provides a means of integrating out the steady state error regardless of operating level or ambient conditions if the system is not power-saturated. However, because of the lag introduced by this type of controller, it may not be possible to meet the rise time requirement.

The velocity plus proportional controller provides zero steady error without compromising rise time to the extent that a pure velocity controller does. This is because the low frequency phase lag introduced by the velocity plus proportional controller is compensated for at higher frequencies by the lead time constant K_p/K_v shown in Table 3. However, this type of controller may, physically, be the most complex to design.

It was observed that, without restricting the turbine speed overshoot to 5 per cent, a large gain was possible even with a pure velocity controller before instability resulted. Thus, an appropriately chosen controller makes possible the satisfaction of a rather stringent set of performance requirements without the introduction of additional forward or feedback loop compensation. This is possible because at low frequencies the system unalterable elements, combined, behave as a single element with a simple first order time constant. This time constant, at any thrust level, is simply the ratio between the turbine inertia and a viscous damping coefficient, which is proportional to thrust level (1). However, an effective viscous damping coefficient may be found, which for purposes of dynamic analysis, adequately defines the

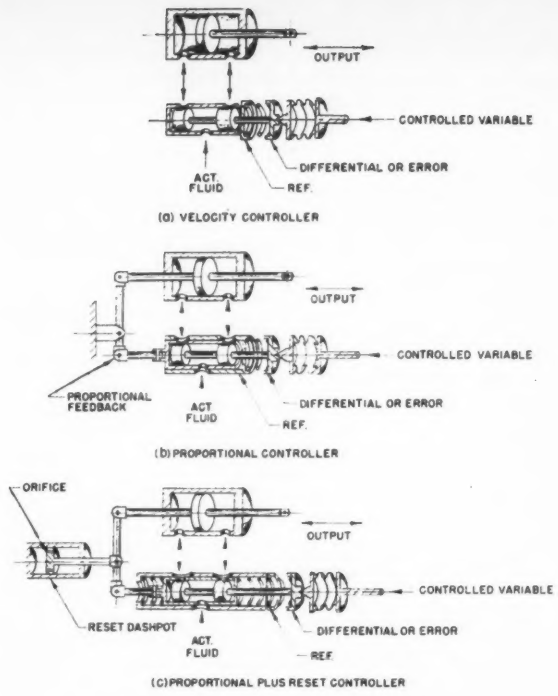

Fig. 10 Basic hydraulic controllers

Table 3 Transfer functions for basic controllers

Controller type	Transfer function for zero controller time constant	Transfer function for finite controller time constant T_c
Velocity ($K_p = 0$)	$\frac{X_a(s)}{\epsilon} = \frac{K_v}{s}$	$X_a(s) = \frac{K_v}{s(T_c s + 1)}$
Proportional ($K_v = 0$)	$\frac{X_a(s)}{\epsilon} = K_p$	$X_a(s) = \frac{K_0}{(T_c s + 1)}$
Proportional plus Velocity ($K_p \neq 0$, $K_v \neq 0$)	$\frac{X_a(s)}{\epsilon} = K_p + \frac{K_v}{s} = \frac{K_v}{s}\left[\frac{K_p}{K_v}s + 1\right]$	$X_a(s) = \frac{K_v(K_p/K_v)s + 1}{s(T_c s + 1)}$

Table 4 Relative controller effectiveness for step responses between 50% and 100% thrust levels

Controller type	Controller time constant, $\%T_b$	Rise time for 5% Overshoot, $\%T_b$	Description of transient
Velocity	0	160	zero steady state error, but relatively sluggish
	20	270	
Proportional	0	50	very rapid rise time, but steady state error due to gain and tolerance limits
	20	60	
Proportional plus velocity	0	72	rapid rise time and zero steady state error
	20	92	

turbopump assembly time constant for operation between 50 and 100 per cent rated thrust.

Fig. 11 shows the simulated step responses of the rocket engine for a change in thrust level from 50 to 100 per cent rated thrust for negligible controller time constant and for various values of K_p and K_v. The optimum values of K_p and K_v, respectively, are designated $\bar{K}_p$ and $\bar{K}_v$. Fig. 11(a) is the step response for $K_p = \bar{K}_p$ and $K_v = \bar{K}_v$. The response is almost critically damped, as is necessitated by a 5 per cent limit on turbine overspeed at the 100 per cent rated thrust level. The time to reach rated thrust is approximately 0.65 T_b.

In Figs. 11(b) and 11(c), K_p is held equal to $\bar{K}_p$, and K_v is varied. In Fig. 11(b), $K_v = 0.15\bar{K}_v$, and, although the initial rise time is rapid, a long period of integration is required to achieve 100 per cent rated thrust because of the low value of K_v. In Fig. 11(c), $K_v = 3\bar{K}_v$ and, although the steady state error is rapidly integrated out, the overshoot is excessive. This overshoot occurs because the system stability margin is reduced by the additional low frequency phase lag introduced by increasing K_v.

In Figs. 11(d) and 11(e), K_v is held equal to $\bar{K}_v$, and K_p is varied. In Fig. 11(d), $K_p = 0.2\bar{K}_p$, and the resultant increase in rise time and overshoot is due to the decrease in the proportionality constant. In Fig. 11(e), $K_p = 4\bar{K}_p$. The rise time is reduced slightly, but the increase in gain causes an undesirable overshoot and oscillation.

Fig. 12 shows the simulated responses of the rocket engine to a change in thrust level from 50 to 100 per cent rated thrust for a nonzero controller time constant. Fig. 12(a) is the same as Fig. 11(a); i.e., the controller time constant is zero, $K_v = \bar{K}_v$, and $K_p = \bar{K}_p$. In Fig. 12(b) the controller time constant T_c is increased to 10 per cent of T_b and a slight amount of overshoot is evident. Even more overshoot occurs, as shown in Fig. 12(c), for $T_c = 20$ per cent T_b. Fig. 12(d) shows that for this value of controller time constant zero overshoot can be restored if K_p is reduced to $0.4\bar{K}_p$. The rise time for 100 per cent rated thrust is 0.90 T_b. As shown in Fig. 12(e) a further increase in controller time constant to 50 per cent T_b is possible before an appreciable overshoot is introduced.

From the transient responses of Figs. 11 and 12, it is possible to conclude:

1 Large K_p, while reducing rise time, introduces undesirable overshoot and an oscillatory response characterized by a relatively high frequency of oscillation.

2 Large K_v, while rapidly reducing steady state error, causes excessive overshoot and an oscillatory response characterized by a relatively low frequency of oscillation.

3 A suitable compromise between K_p and K_v can be achieved which provides zero steady state error, a rapid rise time (less than T_b) and less than 5 per cent overshoot over a reasonably large range of values of controller time constant.

Regardless of the controller type, the transient responses between 100 and 50 per cent rated thrust are more oscillatory than those between 50 and 100 per cent rated thrust. This is due partially to the increase in turbopump assembly time constant at the lower thrust levels, but principally to the increase in the intrinsic system loop gain. For most applications, however, since undershoot in turbine speed is not critical and is not necessarily excessive at the lower thrust levels, it is possible to establish the velocity and proportionality constants K_v and K_p on the basis of the transient between 50 and 100 per cent rated thrust. If necessary, the controller gain can be made proportional to thrust level. Fig. 13 is a typical record illustrating the transient responses in chamber pressure, turbine speed, and controller actuator position for a commanded step change in thrust from 100 per cent rated thrust to 50 per cent rated thrust and from 50 back to 100.

Conclusions

The principal dynamic characteristics of a pump fed, variable thrust, bipropellant, liquid rocket engine system operat-

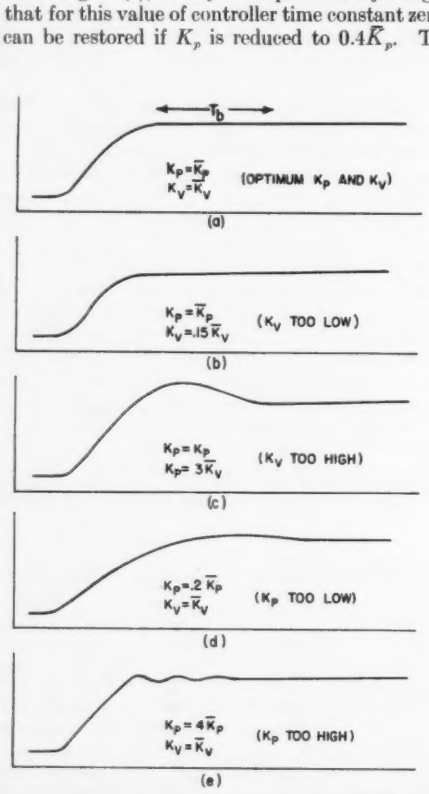

Fig. 11 Step response in chamber pressure of rocket engine to a change of thrust level from 50 to 100 per cent rated thrust with a proportional plus reset controller without time lag

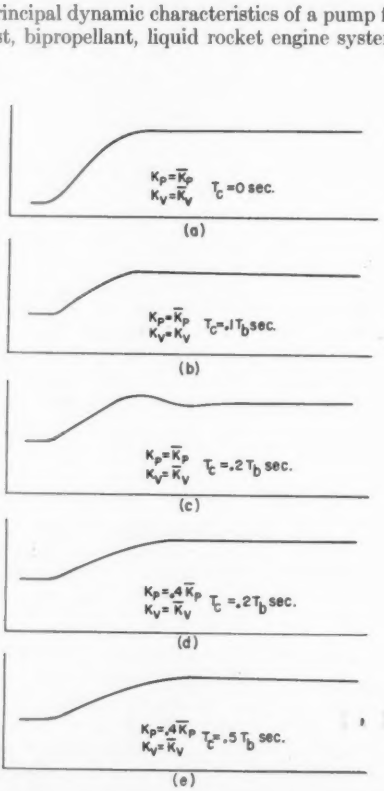

Fig. 12 Step response in chamber pressure of rocket engine to a change of thrust from 50 to 100 per cent rated thrust with a proportional plus reset control with time lag

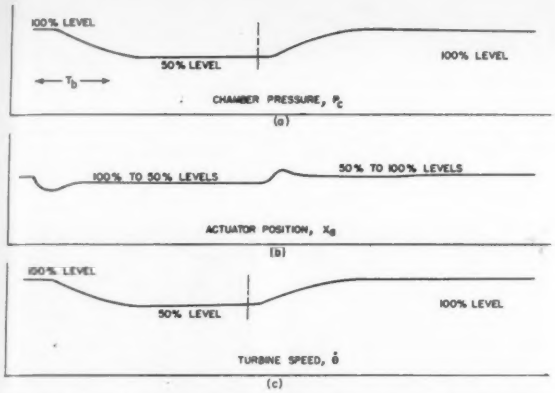

Fig. 13 Transient response of rocket engine system between 50 and 100 per cent rated thrust levels with proportional plus reset controller

ing over a broad range of thrust and meeting a stringent set of performance requirements are summarized below:

1 A variable thrust, bipropellant, liquid rocket engine system is composed of two major units:

(a) The unalterable elements, such as lines, turbine, pumps, and thrust chamber, which, to a large extent, must be designed on the basis of steady state thrust requirements and which must provide static stability.

(b) The alterable, or control elements which must be designed to meet specified transient performance specifications and which must provide adequate dynamic stability.

2 The unalterable elements contain an intrinsic closed loop (i.e., the propellant line-thrust chamber loop) which, because of the low pass filter characteristic of the turbopump assembly, is relatively unaffected by the outer, or control loop, but which:

(a) Must be stable.

(b) Introduces high frequency phase lag into the control loop, and hence, limits the control system loop gain at any thrust level.

(c) Introduces a change of gain, which is inversely proportional to operating level, into the control loop.

3 The decrease in stability of the complete control system at lower thrust levels is due principally to the change in closed loop gain of the propellant line-thrust chamber loop rather than to changes in frequency variant parameters. The nonlinear equations, linearized at any thrust level, may be extended to other thrust levels by calculating the effective open loop gain at the new thrust level.

4 Complex compensation for system dynamics is not necessary because, over the frequency spectrum of importance to the power control loop, the combined unalterable elements appear as a single element with a simple first order time constant which is the ratio between turbopump assembly inertia and an effective damping coefficient.

5 A velocity plus proportional controller is capable of meeting a stringent set of response time, overshoot, and steady state error requirements. Furthermore, the values of the proportionality and velocity constants are not critical functions of specified response time, steady state error, and overshoot; of controller time constant; or of thrust level.

6 The results of this study are of general applicability to any rocket engine power control system, either fixed or variable thrust, of similar nature.

Acknowledgments

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Stabilization of Premixed Propane-Air Flames

(*Continued from page 34*)

type flameholders is that the flame will not cover the cross-sectional area of the duct unless the latter is very long. This disadvantage could be overcome by incorporating a recess around a streamlined body placed in the flow stream creating additional flame fronts. Pressure losses with this arrangement would still be low. The wall recess method of stabilization does offer certain possibilities in laboratory studies. Thus, one may study conveniently the effects of recirculation volume, transport area, edge conditions, and boundary layer characteristics. Also, because the metal walls are good heat conductors and because the flame is in constant contact with them, the flameholder itself is self-heating, thus contributing to performance.

The afore-mentioned studies seem to indicate that the general shape or contour of the recess in the wall is not too important as far as flame stability is concerned, provided the recess occurs abruptly enough to cause separation in the flow stream and the recirculation volume is of sufficient size and of correct proportions.

Acknowledgments

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The Structure of Chlorine Flames

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As a preliminary toward a better understanding of the effect of chlorine compounds on combustion, flames have been burned using chlorine as the sole oxidant. The structure, temperature, and spectral emission of hydrogen- and hydrocarbon-chlorine flames have been studied. The most striking feature of the hydrocarbon-chlorine flame is the large quantity of soot formed in the flame. This carbon formation leads to very high heat losses through radiation, which in turn affect not only the temperature, but also the propagation of the flame.

Introduction

IT IS common to find references in textbooks on inorganic chemistry to the fact that organic liquids will burn in chlorine producing large amounts of soot, and it is well known that mixtures of hydrogen and chlorine are explosive. There have been many studies of the slow reaction between hydrogen and chlorine (1),[3] which is one of the classic examples of a chain reaction, and both the photochemical and thermal chlorination of hydrocarbons have been examined (2), but there have been no systematic studies of chlorine flames.

Flemes supported by fluorine and chlorine trifluoride have recently been studied (4) and these are extremely vigorous flames, both with hydrogen and most organic and inorganic fuels. On the other hand, bromine will only burn with hydrogen and not with hydrocarbons (5), although it might burn with acetylene (6). Chlorine occupies the intermediate position between fluorine and bromine in the halogen group of elements, and hence its behavior as an oxidant is of particular interest.

The behavior of chlorine in flames is also of interest for another reason. Halogenated hydrocarbons are widely used as fire-extinguishing agents and it has recently been shown (5) that the action of methyl bromide can be explained in terms of the addition of an inhibitor (bromine) and additional fuel (CH_3); to understand the action of compounds such as chlorobromomethane and carbon tetrachloride, it will probably be necessary to also know the effect of chlorine on flames.

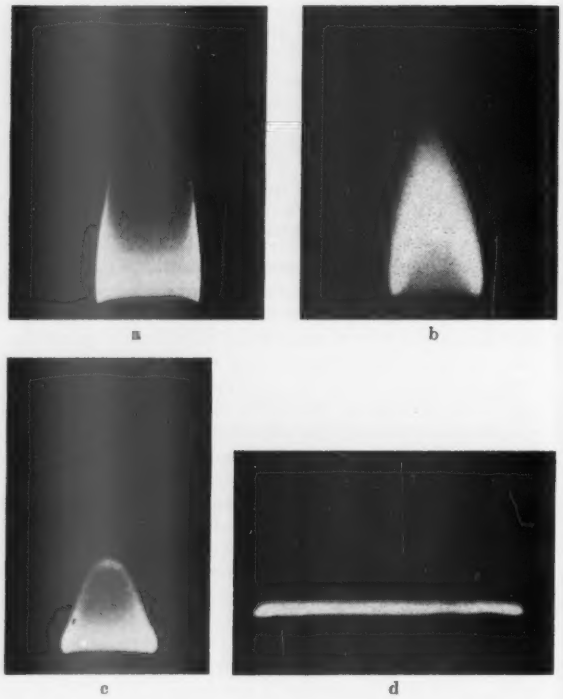

Fig. 1 Chlorine flames. (a) Ethane-air diffusion flame in which chlorine has been added to the ethane. Selochrome plate; f8; $^1/_{50}$ sec exposure. (b) Ethane-chlorine premixed flame (ethane rich). HP3 plate; f8; 1 sec exposure. (c) Stoichiometric ethane-chlorine premixed flame illuminated to show the soot formation. Selochrome plate; f8; $^1/_4$ sec exposure. (d) Flat stoichiometric ethane-chlorine premixed flame. Selochrome plate; f8; $^1/_{50}$ sec exposure

Results

Description of Premixed Chlorine Flames

When chlorine is added to a hydrocarbon burning as a diffusion flame in the surrounding air, on a 9-mm-diam burner, sooting becomes quite pronounced; with the addition of sufficient chlorine, the hydrocarbon-air diffusion flame strips off, leaving behind a dull orange premixed hydrocarbon-chlorine flame which had previously been hidden by the outer diffusion flame. This change is illustrated in Fig. 1 (a, b). At this point the premixed flame is fuel rich; as further chlorine is added the yellow flame becomes much brighter, and considerably greater quantities of soot are formed. The light emission of this flame is entirely due to radiating carbon particles. Both the brightness of the reaction zone and the formation of soot reach a maximum with a mixture which contains slightly more chlorine than that required to give only hydrogen chloride and soot as products. With this flame, shown in Fig. 1(c), the burned gases cool very rapidly, and quite a short distance above the bright yellow reaction zone only black carbon can be seen. With very chlorine rich flames there is no carbon formation and the reaction zone changes from yellow to blue-grey; this coloration extends well into the burned gases with very little loss in intensity.

Burning velocity measurements have been made with ethane-chlorine mixtures by the Bunsen cone method, but the values obtained are only approximate as it was difficult to decide the exact position of the reaction zone. Measurements were made over a range of mixture composition and the value obtained was between 5–10 cm/sec in all cases, the most reliable values being around 5 cm/sec.

Premixed ethylene-chlorine flames could be burned, despite the slow reaction between ethylene and chlorine which occurs during the mixing period. Acetylene and chlorine, on the other hand, react sufficiently rapidly to prevent a study of the premixed flame due to "flashback." In this respect it is

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[3] Numbers in parentheses indicate References at end of paper.

interesting to note that a mixture of acetylene, air, and chlorine is self-igniting at atmospheric pressure.

Hydrogen will burn very vigorously with chlorine, giving a pale blue-grey flame; Bartholomé (3) has measured the burning velocity of these flames. He found that the maximum burning velocity was just over 400 cm/sec and that this occurred with very fuel rich mixtures.

The effect on the burning velocity of replacing the air of an ethane-air flame by chlorine is shown in Fig. 2. It will be seen that the first addition of chlorine results in a rapid decrease in burning velocity, but that on further replacement the burning velocity does not change very appreciably.

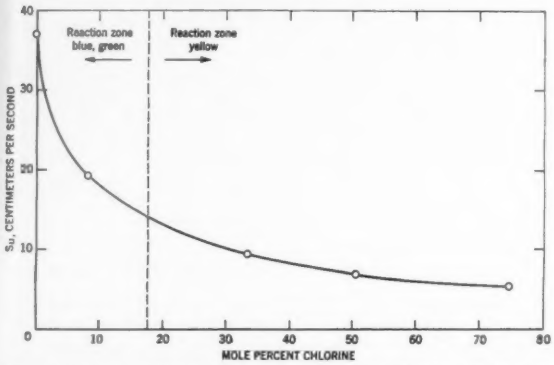

Fig. 2 The burning velocity of ethane-air-chlorine mixtures; burner diam 9 mm—all mixtures examined lie on the line *AB* in Fig. 4

Limits of Inflammability of Fuel-Air-Chlorine Systems

Limits of inflammability were determined under standard conditions of 50 C and 380 mm Hg by the method described previously (5). In this way a direct comparison could be made between the effect of bromine and chlorine. This method uses visual observation to determine whether or not a flame propagates through the mixture, but this is not possible with the hydrogen-air-chlorine system, since the flames were both very faint and fast. Instead, the pressure kick and the change in final pressure which accompanies flame propagation were used to detect flame propagation.

Fig. 3 shows the limits of inflammability of the systems hydrogen-air-chlorine and carbon monoxide-air-chlorine, and it

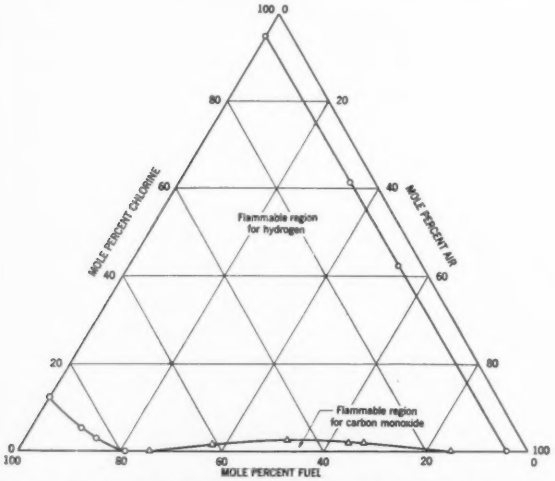

Fig. 3 Limits of inflammability of the systems hydrogen-air-chlorine and carbon monoxide-air-chlorine

Temp = 50 C
O = hydrogen
Pressure = 380 mm Hg
△ = carbon monoxide

NOTE: The carbon monoxide contains 2% hydrogen.

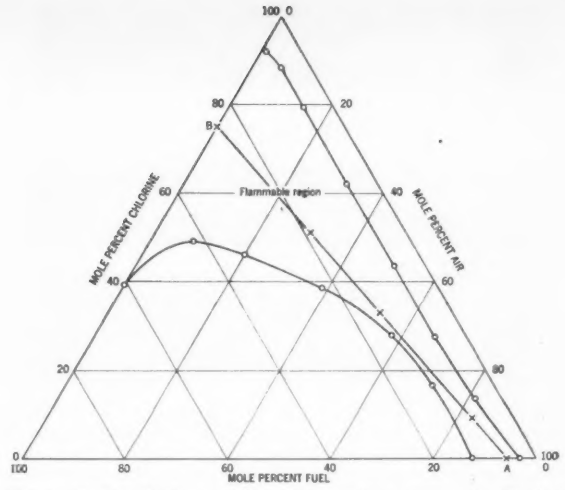

Fig. 4 Limits of inflammability of the system ethane-air-chlorine

Temp = 50 C
O = limits of inflammability
Pressure = 380 mm Hg
× = points for which burning velocities were determined (see Fig. 2)

will be seen that whereas chlorine has no effect on the hydrogen-air system it has a very pronounced inhibiting effect on carbon monoxide–air. The lean and rich hydrogen limits are approximately straight lines and thus follow Le Chatelier's mixing rule. It is clear, therefore, that in contrast to the marked inhibiting effect of bromine on hydrogen-air, chlorine has no inhibiting action. With carbon monoxide–air, on the other hand, chlorine is a very efficient inhibitor, concentrations of the order of only 2.5 per cent being sufficient to prevent flame propagation. Attempts were made to ignite a 1:1 mixture of carbon monoxide and chlorine, as this is the most probable mixture to support a flame by forming phosgene. This and similar mixtures could not be ignited.

The limits of inflammability of the system ethane-air-chlorine are shown in Fig. 4. It will be seen that the lean ethane limit is approximately a straight line, as with the hydrogen system; but this is not the case with the rich ethane limit. The addition of air has a marked inhibiting effect on the rich ethane-chlorine limit. It is interesting to note that oxygen strongly inhibits the high temperature thermal chlorination of ethane (7).

Spectrographic Observations

Premixed Flames

All spectra, both of premixed and diffusion flames, were taken with a Hilger medium quartz spectrograph. For the study of premixed flames a burner giving a flat flame with a good optical depth (approximately 3 cm) was used, and an image of the flat flame was projected onto the vertical slit of the spectrograph. A drilled plate burner was used, in which the burner plate consisted of a solid brass disc with $^1/_{32}$ in. diam holes spaced $^1/_{16}$ in. apart at the corners of a series of equivalent triangles. The flow velocities were so small that a nearly perfect flat flame front was formed a few millimeters above the burner. A photograph of this flame is shown in Fig. 1(d).

It will be seen from Fig. 5(a) that hydrocarbon-chlorine flames only emit continuous radiation. With both very lean and rich mixtures the emission is due to continuous gas radiation and resembles that of the hydrogen-chlorine flame, but for mixtures between these two extremes the radiation is due to carbon particles, and thus closely resembles black-body radiation. No bands due to C_2 or CH radicals were detected. The possibility that this is due to the fact that the band emission

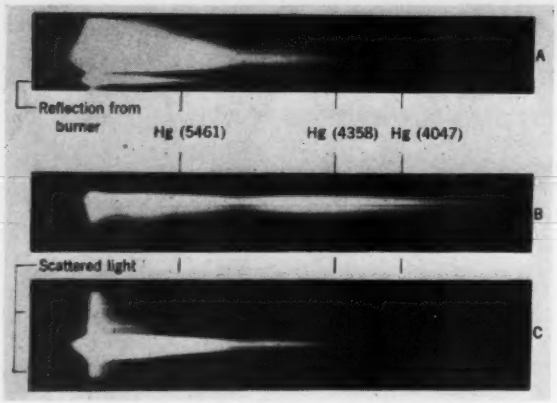

Fig. 5 Spectra of chlorine flames—all spectra taken with a medium Hilger quartz spectrograph. (a) Premixed ethylene-chlorine flame (8.8 cc/sec C_2H_4, 25 cc/sec Cl_2). HP3 plate; slit 0.05 mm; exposure time 6 min. (b) Flat diffusion flame of hydrogen and chlorine. HP3 plate; slit 0.05 mm; exposure time 3 min. (c) Flat diffusion flame of ethylene and chlorine. HP3 plate; slit 0.05 mm; exposure time 3 min

is masked by the black-body radiation of carbon particles can be rejected since, as will be seen later, the emissivity is not unity at the beginning of the reaction zone. It is from this position that radiation from C_2 or CH radicals is usually emitted, and hence it is clear that these radicals are not formed during the combustion process.

Diffusion Flames

Diffusion flames were burned on a flat flame burner (8) which consisted of two adjoining rectangular slots (1 cm × 3 cm) carrying the fuel and chlorine, respectively; a cross section through the flame, about 0.5 cm above the interface, was studied. An image of the end view of the flame was focused onto the horizontal slit of a Hilger medium quartz spectrograph, and in this way the structure of the flame could be examined.

The spectrum of a hydrogen-chlorine diffusion flame is shown in Fig. 5(b); it will be seen to consist mainly of a continuum in the visible and near ultraviolet. This spectrum is presumably identical with that of the premixed hydrogen-chlorine flame. Urey and Bates (9) have suggested that this continuum is due to the recombination of chlorine atoms. (See also Palmer (10) and Kitagawa (11).)

The spectrum of an ethylene-chlorine flame is shown in Fig. 5(c). It only consists of radiation from carbon particles, and this is the case for all hydrocarbon fuels. The difference of the intensity distribution of the continua in Fig. 5(b,c) is especially noticeable.

Temperature Measurements

Temperature measurements were made on flat premixed flames burned on the drilled plate burner described above. The flames on this burner had a bright yellow flat zone emitting radiation from carbon particles which cooled down very rapidly, giving the impression of a luminous reaction zone. This is illustrated in Fig. 1(d).

A slightly enlarged image of the flat flame was projected onto the vertical slit of a Hilger medium quartz spectrograph, the slit length of which, however, was restricted to 0.75 mm so that only a small part of the flame was investigated at a given time. A series of positions across the reaction zone could be investigated by moving the flame up and down with a screw arrangement. The intensity of the light passing through the slit was measured with a photomultiplier in the focal plane of the spectrograph with a fixed wave-length position at 5700 Å. The slit width for both the spectrograph and the photomultiplier was 0.1 mm. The image of a tungsten

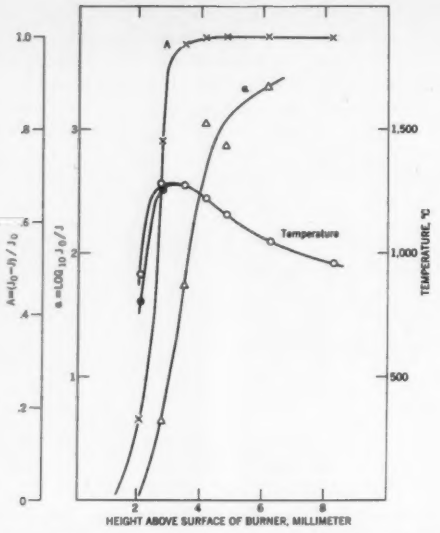

Fig. 6 Measurement of the absorption of light and of the temperature in a flat premixed ethane-chlorine flame

× = absorptivity
△ = extinction coefficient
○ = true temperature
● = brightness temperature

filament lamp could also be projected into the flame, and hence also onto the spectrograph slit. In this way the photomultiplier could be calibrated against the tungsten lamp.

The experimental procedure was to measure the photomultiplier output for three arrangements:

(a) Tungsten filament lamp alone at a higher temperature than that of the flame.
(b) Tungsten lamp as in (a) but shining through the flame.
(c) Flame alone.

In addition, the photomultiplier output was calibrated against the brightness temperature of the tungsten lamp. These measurements enabled the absorptivity (A) and the brightness temperature (T_{Br}) of the flame to be measured for a given position in the flame. The absorptivity of the flame is defined as

$$A = \frac{J_0 - J}{J_0}$$

where J_0 is the radiation from the lamp and J the radiation from the lamp after passing through the flame, i.e., the difference between the measurements (b) and (c). The brightness temperature could be obtained by direct comparison with the tungsten lamp and the true temperature (T_{Tr}) from T_{Br} and A.

The absorptivity depends on, but is not proportional to, the amount of carbon formed. A more suitable quantity for measuring the quantity of carbon is the extinction coefficient (α), where

$$J = J_0 10^{-\alpha}$$

i.e.

$$\alpha = \log (J_0/J)$$

Even with the extinction coefficient, however, there is the disadvantage that it may depend on the size distribution of the carbon particles as well as the total mass of carbon in the flame, and nothing is known about this size distribution.

The results of the measurements on a flat ethane-chlorine premixed flame are shown in Fig. 6. The abscissa represents the position in the flame, as measured by the height above the burner surface; the ordinate represents the absorptivity, ex-

tinction coefficient, brightness, and true temperature of the flame. It will be seen that except for the very beginning of the flame the absorptivity is very close to unity for all positions in the flame—close enough to render the difference between true and brightness temperature negligible.

The first point on the temperature curve (T_{Br} = 800 C) is not very accurate due to the very small value of the photomultiplier output, but the curve shows the rapid increase in temperature just in front of the flame front and the decrease behind the flame front. It is interesting to note that the increase in α lags behind both the temperature increase and the rapid increase in absorptivity. Unfortunately the large values of α are liable to be inaccurate since J, the radiation of the lamp going through the flame, is the difference of two large values, i.e., the radiation from both flame and lamp, and flame alone.

Fig. 7 shows a similar curve for a flat ethylene-chlorine flame. The maximum temperature of this flame is slightly lower than that for ethane-chlorine, and, as in the case of the ethane-chlorine flame, carbon formation is slightly delayed in comparison with the increase in temperature.

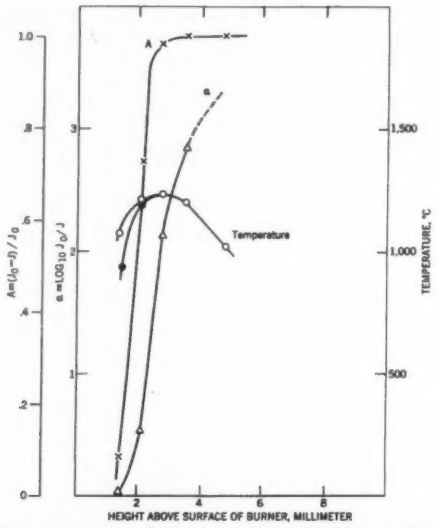

Fig. 7 Measurement of the absorption of light and of the temperature in a flat premixed ethylene-chlorine flame

× = absorptivity
△ = extinction coefficient
○ = true temperature
● = brightness temperature

Discussion

Hydrogen Flames

From Fig. 3 it will be seen that Le Chatelier's mixing rule holds for the hydrogen-air-chlorine system, and thus the chlorine-hydrogen and hydrogen-air systems do not interact on each other. Lindeijer (12) has shown that this is also true for the system hydrogen-oxygen-chlorine. This behavior is in marked contrast to that of bromine which has a pronounced inhibiting effect on the limits of inflammability of hydrogen-air mixtures (5).

The main chain propagation reactions in the hydrogen-air system are

$$OH + H_2 = H_2O + H \quad [1]$$

$$H + O_2 = OH + O \quad [2]$$

$$O + H_2 = OH + H \quad [3]$$

and since reaction [2] has the highest activation energy it is the rate controlling step in the over-all reaction. When a halogen is also introduced into the system, hydrogen atoms are removed by reaction [4] and halogen atoms formed

$$H + X_2 = X + HX \quad [4]$$

Both bromine and chlorine atoms are capable of continuing the chain process by reacting with hydrogen molecules, but whereas the reaction of chlorine atoms with hydrogen is effectively thermoneutral with a low activation energy, the reaction of bromine atoms requires an activation energy of 17.7 kcal and is endothermic to the extent of 16.7 kcal. In consequence, it seems quite probable that the inhibiting action of bromine is due to the fact that unreactive bromine atoms are formed which slow down the reaction rate sufficiently to prevent chain propagation. Further work to confirm these ideas is very desirable.

In contrast to the effect of chlorine on the hydrogen-air system, it strongly inhibits both carbon monoxide–air and carbon monoxide–oxygen flames. This inhibition cannot be explained at the moment.

Hydrocarbon Flames

The stoichiometric ethane-air flame is bright yellow and large quantities of soot are produced. In contrast, fuel rich flames, near the rich limit, are dull orange and less carbon is produced than in the stoichiometric flame. Similarly, with lean mixtures no carbon is formed at all, and the color of the flame, which extends right into the burned gases, is blue-grey. The temperature of fuel rich flames is quite low and this seems to prevent the decomposition of excess fuel. Again, at the lean limit the flame temperature is low; in consequence, no carbon is formed, as carbon-chlorine compounds are stable. This latter point was confirmed by taking a gas sample of the products from the fuel lean flame; a solid crystallized out on the walls of the sampling vessel over a period of time and its melting point corresponded to hexa-chlor-ethane (C_2Cl_6). Thus it seems probable that with fuel rich and stoichiometric mixtures hydrogen abstraction reactions predominate leading to carbon formation. With fuel lean mixtures, on the other hand, substitution reactions become important.

Temperature Measurements

As already pointed out, carbon is the main product from hydrocarbon-chlorine flames, and to a very large extent this determines both the stability and temperature of these flames. The radiation from carbon particles was found to be black-body radiation as the absorptivity approached unity. The temperature profile of the flame could be measured and the formation of carbon followed by determining the extinction coefficient. The maximum temperature occurs before the extinction coefficient reaches its maximum value and this may be due to one of two reasons. The first is that cooling due to radiation is so large and occurs so early in the reaction zone that the temperature begins to decrease before all the carbon is produced, i.e., before the reaction in the flame is complete. The second possibility is that the extinction coefficient is bigger for the larger particles, although the total amount of carbon is unaltered.

The adiabatic flame temperature of a stoichiometric ethylene-chlorine flame is about 2000 C, but it will be seen from Fig. 7 that the highest temperature actually measured was only 1250 C and that this was for a chlorine rich mixture. For a stoichiometric mixture the temperature was as low as 1000 C. This suggests that the heat losses from hydrocarbon-chlorine flames are very much higher than usual. This result is also indicated by the rapid cooling of the products immediately after leaving the reaction zone.

Heat losses by radiation do not only occur in a lateral direction, but also in a direction normal to the flame front. The lateral heat losses could be minimized by having a larger diameter flame, but this would not influence the relative losses normal to the flame front. These latter heat losses cannot be calculated because the thickness of the carbon layer which

produces black-body radiation is unknown, and this is necessary for the calculation as the temperature changes rapidly behind the flame front. This conception of large heat losses due to radiation introduces an extra factor which must be taken into account when problems of flame stability are being considered. The light which the flame radiates onto the unburned hydrocarbon-chlorine mixture will be partially absorbed by chlorine and no doubt will influence the flame. It is interesting to note in this connection that hydrocarbon-chlorine flames burning on a glass burner will flash back, when a photoflash lamp is switched on close to the flames.

As pointed out earlier, the burning velocities of hydrocarbon-chlorine mixtures are very small. This is no doubt also due to the high heat losses from the flame and the resultant low temperature. No further comment will, in consequence, be made on these burning velocities.

Spectrographic Observations

In the spectra from premixed hydrocarbon-chlorine flames, in contrast to oxygen or air flames, no bands due to C_2 or CH radicals were detected. The only other hydrocarbon flame for which this has also been observed is the self-decomposition flame of acetylene, with which carbon formation is also the main feature (13). It should be noted that most rich hydrocarbon-air flames, at the point at which carbon is just beginning to form, either emit very weak C_2 bands or none at all. It is obvious that this is not due to the low temperature of these very rich flames since C_2 bands are detected in stoichiometric flames diluted with nitrogen. This might suggest that oxygen is necessary for the formation of C_2 radicals, but it has been found that mixtures of hydrogen and hydrocarbons burning with chlorine trifluoride[4] in a diffusion flame emit very strong C_2 bands in the absence of oxygen (4); in consequence, this suggestion can be rejected. Oxygen, however, must be present for CH radicals to be formed in fluorine flames. The most probable reaction (4) forming CH is

$$C_2 + OH = CH + CO$$

and this may account for the absence of CH radicals in hydrocarbon-chlorine flames. It seems probable that the carbon formation in chlorine flames is a result of a simultaneous stripping of the hydrogen to form hydrogen chloride and a polymerization reaction, without the intermediate formation of small organic fragments which are probably formed in oxygen and air flames. These fragments may result in the formation of C_2 radicals. Before any other conclusions can be drawn about reactions forming and destroying C_2 and CH radicals in flames, it is necessary to have the evidence from a study of premixed hydrocarbon-chlorine trifluoride flames.

As with premixed flames, hydrocarbon-chlorine diffusion flames emit only radiation from carbon particles. Hydrocarbon-oxygen flames split up into two zones, a main reaction zone and a carbon zone in which the fuel is decomposed by the heat from the main reaction zone. In the chlorine flame there is only one single reaction zone in which carbon is formed due to reaction of the hydrocarbon with the chlorine. The absence of any carbon formation due to decomposition of the fuel can probably be attributed to the low temperature of the flame and the great ease with which carbon is formed in the presence of chlorine. The density of the carbon cloud will also help to delay diffusion processes. In this respect the flame resembles hydrocarbon-chlorine trifluoride diffusion flames, which also have only a single carbon zone with no band emission.

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[4] Pure fluorine should behave in the same way, but commercial fluorine usually contains a little oxygen which changes the light emission appreciably.

Flame Stabilization in a Boundary Layer

(*Continued from page 80*)

Acknowledgment

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Just a reminder!

Next month

JET PROPULSION

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Second Annual Special Supplement on

CAREERS IN THE ROCKET AND MISSILE INDUSTRY

Technical Notes

A Simple Equation for Rapid Estimation of Rocket Nozzle Convective Heat Transfer Coefficients[1]

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Although recent analyses of the heat transfer in convergent-divergent nozzles based on considerations of the turbulent boundary layer have appeared in the literature (References 1, 2), the methods require considerable computational time for each new nozzle and have not as yet been confirmed (or denied) by reliable experimental data. This note was prepared because there still appears to be a need for a simple, yet reasonably accurate, approximation equation for making rapid preliminary estimates of the convective heat transfer coefficients in rocket nozzles.

Nomenclature

A = local cross-sectional area of flow
C_p = specific heat at constant pressure, Btu/lb, °F
C = constant coefficient, Equation [6], dimensionless
C^* = characteristic velocity, fps
D = diameter, in.
g = gravitational acceleration, fps²
h_g = heat transfer coefficient, Btu/in.² sec, °F
k = thermal conductivity
m = average molecular weight of combustion gases
M = Mach number
Nu = Nusselt number = $h_g D/k$
p_c = chamber pressure, lb/in.²
Pr = Prandtl number = $\mu C_p/k$
q = heat flux
r_c = throat radius of curvature
Re = Reynolds number = $\rho' UD/\mu$
T = static temperature, °R
T_c = chamber or flame temperature, °R
T_0 = stagnation temperature, °R
x = distance from inlet measured along wall
x_n = distance through nozzle measured along wall
U = free stream value of local gas velocity
γ = ratio of specific heats, dimensionless
μ = viscosity, lb/in. sec
ρ' = free stream value of local gas density
σ = dimensionless factor accounting for variation of ρ and μ values across boundary layer
ω = temperature exponent of viscosity equation

Subscripts

am = arithmetic mean
0 = stagnation conditions
$*$ = conditions at nozzle throat
w = wall

EXPERIENCE gained from the turbulent boundary layer calculation methods (1, 2)[3] has shown that under certain conditions the dominant variable factor is the mass flow rate per unit area and that variations in velocity and temperature boundary layer thicknesses exert only a secondary, although not negligible, effect. This suggests that an equation of the form

$$h_g \sim (\rho' U)^m \quad \ldots\ldots [1]$$

might serve as the desired rough approximation equation. Equation [1] can account for mass flow rate variations by proper selection of m, while it ignores the effect of boundary layer development. (This is not to say that assumption of fully developed pipe flow has been made.) Equation [1] can be put into the more familiar nondimensional form

$$Nu = C(Re)^m(Pr)^n \quad \ldots\ldots [2]$$

which happens to be the same form as the equation that has been used for correlation of fully developed turbulent pipe flow heat transfer data.

From the equation for h_g in (2) the proper value for m can be shown to be 0.8. (Note that $(1/\theta)^{1/4}$ in Equation [31] of (2) is proportional to $Re^{0.05}$.) The value for n is arbitrarily selected as that frequently used for modifications of Reynolds analogy for Pr near unity 0.4. The problem of where to evaluate the physical properties must be answered, since in general large temperature differences are present in rocket nozzles. Since the procedure of evaluating properties at the arithmetic mean (am) between bulk temperature T and wall temperature T_w has been successful for low speed, high temperature difference problems (3) and high speed, low temperature difference problems (4), it is employed here. With these assumptions, Equation [2] can be solved for h_g to give

$$h_g = \frac{C}{D^{0.2}} \left(\frac{\mu^{0.2} C_p}{Pr^{0.6}} \right)_{am} (\rho_{am} U)^{0.8} \quad \ldots\ldots [3]$$

If it is allowed that C_p and Pr do not vary appreciably with temperature, they can be assumed constant at stagnation temperature values, while μ_{am} and ρ_{am} can be evaluated in terms of the stagnation and static temperature values, respectively. Thus Equation [3] can be written

$$h_g = \frac{C}{D^{0.2}} \left(\frac{\mu^{0.2} C_p}{Pr^{0.6}} \right)_0 (\rho' U)^{0.8} \sigma \quad \ldots\ldots [4]$$

where $\sigma \equiv (\rho_{am}/\rho')^{0.8} (\mu_{am}/\mu_0)^{0.2}$. The factor σ contains all the corrections for property variation across the boundary layer. Noting that $T_{am} = 1/2(T + T_w)$, that $\rho \sim (1/T)$, and that $\mu \sim T^\omega$, the value of σ can be evaluated in terms of T_0, T_w, and M. (T_0 is selected since T_0 is constant through the nozzle.)

$$\sigma = \frac{1}{\left[\frac{1}{2} \frac{T_w}{T_0} \left(1 + \frac{\gamma - 1}{2} M^2 \right) + \frac{1}{2} \right]^{0.8 - (\omega/5)} \left[1 + \frac{\gamma - 1}{2} M^2 \right]^{\omega/5}} \quad \ldots\ldots [5]$$

The equation for h_g can be put in a form more easily used for rocket nozzle computations by evaluating $\rho' U$ in terms of C^*, and A_*/A

$$h_g = \left[\frac{C}{D_*^{0.2}} \left(\frac{\mu^{0.2} C_p}{Pr^{0.6}} \right)_0 \left(\frac{p_c g}{C^*} \right)^{0.8} \right] \left(\frac{A_*}{A} \right)^{0.9} \sigma \quad \ldots\ldots [6]$$

Note that the factor in the brackets is a constant through a nozzle leaving only A_*/A and σ to be evaluated at each station.

The value of C was determined by equating the value of h_g computed from Equation [6] at the throat of a particular rocket nozzle with the value computed from the turbulent

[1] This paper presents one phase of research carried out at the Jet Propulsion Laboratory, California Institute of Technology, under Contract No. DA-04-495-Ord 18, sponsored by the Department of the Army, Ordnance Corps.
[2] Research Group Supervisor.
[3] Numbers in parentheses indicate References at end of paper.

EDITOR'S NOTE: This section of JET PROPULSION is open to short manuscripts describing new developments or offering comments on papers previously published. Such manuscripts are published without editorial review, usually within two months of the date of receipt. Requirements as to style are the same as for regular contributions (see masthead page of this issue).

boundary layer analysis (see sample calculation, Reference 2, case 2a) for the same values of mass flow and gas properties. The resulting value of C was found to be 0.026 which is, coincidentally, quite close to the value usually used in heat transfer correlation for turbulent flow in pipes. The agreement between the results of Equation [6] and those of the turbulent boundary layer method over the rest of the nozzle is exhibited in Fig. 1.

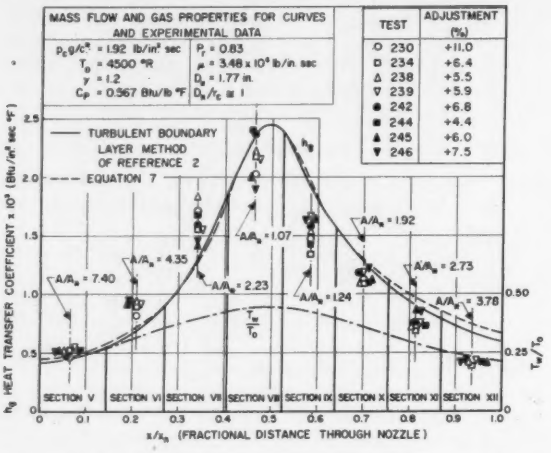

Fig. 1 Distribution of heat transfer coefficient for nozzle of Fig. 2, Ref. 2

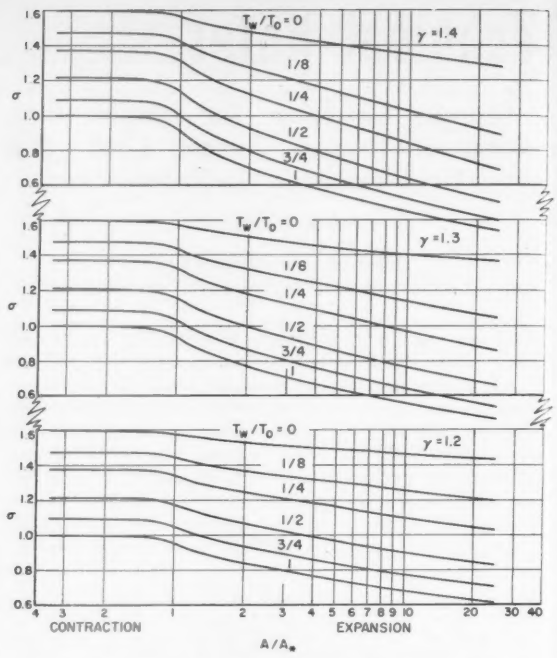

Fig. 2 Values of the properties variation parameter σ

In writing the equation in its final form, a factor to account for the effect of radius of curvature of the nozzle throat r_c is added. This factor, $(D_*/r_c)^{0.1}$ was suggested by the nozzle similarity considerations of (2)

$$h_g = \left[\frac{0.026}{D_*^{0.2}}\left(\frac{\mu^{0.2}C_p}{Pr^{0.6}}\right)_0\left(\frac{p_c g}{C^*}\right)^{0.8}\left(\frac{D_*}{r_c}\right)^{0.1}\right]\left(\frac{A_*}{A}\right)^{0.9}\sigma \quad \ldots\ldots [7]$$

The particular nozzle contour for which the comparison between the simplified equation of this paper and the turbulent boundary layer method was made was one with D_*/r_c equal to about unity and having contraction and expansion half angles of 30 and 15 deg, respectively. Although nozzles with different angles will probably not show the same agreement between methods, the simplified equation will probably be sufficiently accurate for its intended purpose if the contraction and expansion angles are not changed by more than 50 per cent and the value of D_*/r_c is not greater than about 3.

As a step toward the goal of a rapid calculation method, values of σ have been computed for γ of 1.2, 1.3, 1.4; for ω of 0.6; and for various values of T_w/T_0. These are plotted in Fig. 2 vs. A/A_* on a log scale increasing on both sides of the minimum value of unity.

If Pr and μ data are not available for the particular combustion-gas mixture under consideration, kinetic theory (5) can be used to get the approximate result

$$Pr = \frac{4\gamma}{9\gamma - 5} \quad \ldots\ldots [8]$$

The NBS (6) data for the viscosity of air at high temperatures can be used to get a correlation equation which should be reasonably accurate for most mixtures consisting principally of diatomic gases.

$$\mu = (46.6 \times 10^{-10})(\bar{m})^{1/2}(T\ ^\circ\mathrm{R})^{\omega}\ \mathrm{lb/in.\ sec} \quad \ldots\ldots [9]$$

where $\omega = 0.60$. The average molecular weight factor is suggested by the statistical mechanical transport property theory (7). The value of (C_p) is usually known from thermochemical calculations for the combustion gases under consideration.

For comparison with experimental h_g data, values measured[4] semilocally in a nozzle previously described in the literature (8) operating with the combustion gases of the RFNA-N_2H_4 system are also plotted in Fig. 1.

The vertical dashed lines about which the data are grouped represent the center lines of the sections defined by the vertical solid lines. Some points are plotted off the line to facilitate identification of individual data points. To achieve a common basis for comparison, these data were all adjusted to the values of mass flow and gas properties listed in the figure. The adjustments were made according to the dependence on gas flow properties indicated by Equation [7]. The maximum adjustment was 11 per cent. The experimental values of h_g were computed from calorimetrically measured semilocal values of q, computing T_0 from theoretical T_c values reduced by the square of the ratio of experimental to theoretical C^*. Wall temperatures T_w were calculated from considerations of water-cooling conditions and temperature drop through the wall.

These experimental data should not be construed as supporting or denying the results of either the equation presented or the turbulent boundary layer method. Additional reliable local data must be obtained for proper experimental evaluation of either method. Rather, it is significant (*a*) that the results of the simple correlation equation developed in this paper agree reasonably well with *a particular set* of calculations based on the turbulent boundary layer development in the nozzle, and (*b*) that both methods show reasonable agreement with the meager experimental data available from rocket motor tests. It must be remembered that the agreement between the two methods near the entrance is strongly dependent on the assumed entrance boundary layer conditions, which in this calculation may or may not have been typical of rocket-motor nozzles.

Less direct experimental verification of the equation presented is offered by the excellent agreement between average values of heat flux over a large contraction ratio nozzle computed using Equation [7] and average heat flux measurements made over a wide range of chamber pressures with the RFNA-NH_3 system (9). This equation was notably unsuc-

[4] By of E. L. Wilson of the Jet Propulsion Laboratory.

cessful in predicting chamber heat fluxes for these tests, the values being only 40–50 per cent of the measured values. This is not altogether surprising since convection related to the average mass flow rate is not the only important mode of heat transfer in a chamber. However, in a nozzle, at least near the throat and beyond, heat fluxes should be successfully predicted by considering convection based on the average mass flow rate, except when (*a*) a substantial fraction of the gases are strong radiators, (*b*) there is substantial dissociation with subsequent recombination near the wall, or (*c*) there are strong high frequency flow instabilities, in which cases the predictions are expected to be too low. Such predictions will be too high when (*a*) the combustion gases deposit insulating solids on the walls and (*b*) the combustion reactions are not completed in the chamber.

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Note on "Hazards Associated With 90 Per Cent Hydrogen Peroxide Aerosols"[1]

CHARLES L. PUNTE

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SOME concern has been expressed about the severe effects of exposure to aerosols (liquid airborne particles) of H_2O_2 reported in the article "Hazards Association With 90 Per Cent Hydrogen Peroxide Aerosols."

It should be pointed out that the aerosol concentrations used in the animal experiments reported are very high and are likely to persist under accidental conditions where a constant jet of material is fed into an enclosed space. The real hazard would exist when personnel were trapped in such a space without adequate protective equipment. Where escape is possible it is likely that the irritating effects to eyes and nose would preclude prolonged exposures to high aerosol concentrations which might be lethal. It is unlikely that high aerosol or liquid airborne particle concentrations would exist under normal handling conditions.

Received Dec. 3, 1956.

[1] By Charles L. Punte, Leon Z. Saunders, and Eugene H. Krackow. JET PROPULSION, vol. 26, June 1956, p. 500.

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TYPICAL OPERATION

	400 cps	500 cps
Coil voltage	6.3V sine, square, pulse wave	6.3V sine, square, pulse wave
Coil current	55 milliamperes	45 milliamperes
Coil resistance	85 ohms	85 ohms
*Phase lag	55° ±10°	65° ±10°
*Dissymmetry	less than 4%	less than 4%
Temperature	−55°C to 100°C	−55°C to 100°C
*Switching time	15° ±5°	15° ±5°

Mounting—Any position—fits 7-pin miniature socket

*These characteristics based on sine wave excitation

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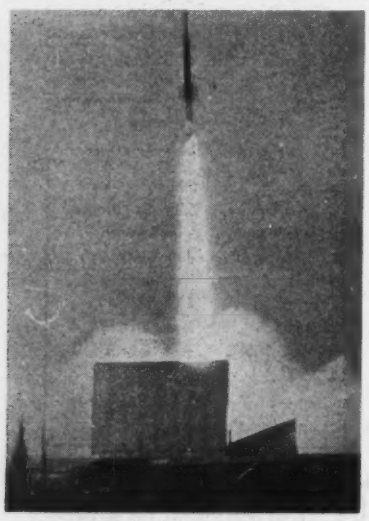
U. S. Army photograph

First to go at Fort Churchill, Nike-Cajun was fired for familiarization.

Churchill Firings Prove It Can Be Done

As a preliminary to the upcoming International Geophysical Year rocket research program, the United States National Committee for IGY recently launched six high altitude sounding rockets from Fort Churchill, Canada, in an attempt to feel out the arctic atmosphere.

Termed a tremendous success, the firings also provided valuable scientific data. Most significant, for example, are the data from a firing instrumented to determine composition of the air and ion structure at high altitude. Although still not completely analyzed, say scientists connected with the project, this data will undoubtedly contribute greatly to the advancement of long-range communication and guidance.



Nike-Cajun on its way.

Not only will this new data extend knowledge of ionospheric penetration (i.e., getting through to rockets above the upper layers), but it is also expected to contribute to improvement in the use of ionospheric reflection, the method by which most signals are bounced over long distances. Conceivably knowledge gleaned from this and similar firings could lead to trans-oceanic television and the like. And, possibly, this data may have some import for interplanetary travel in regard to the possible ionization effect on space vehicles.

Readings from the other rockets fired at Fort Churchill—most of which were instrumented for pressure, temperature, and density experiments—are also expected to make significant contributions to the knowledge of the upper atmosphere, says Robert M. Slavin of the Air Force Cambridge Research Center which launched two rockets in the series. But, principally, the preliminary firings served to check out range and facilities at Fort Churchill for the forthcoming IGY. Secondarily, but almost as significant, this joint effort of the U. S. National Committee for IGY and the Department of Defense showed that with the high degree of cooperation achieved among the three services and group of civilian scientists a major effort in basic research could be made on a very limited budget.

The tests, of course, were conducted by the civilian scientists. But establishment and running of the over-all facility was carried out by the Army, Navy and Air Force under direction of a Defense Department Interservice Coordinating Group.

Trouble in the North: And, most important, says John W. Townsend who is in charge of the Naval Research Laboratory field group at Churchill, these firings proved it can be done. "We've learned that we can get payloads up to predicted altitudes under arctic conditions." This was no mean feat in itself.

Air and water lines froze. Power failures were frequent. Radar equipment was beset with cold weather problems. Holes were blasted through the specially designed, enclosed launching tower and it became impossible to maintain temperatures.

And, unlike White Sands and Holloman, weather conditions and terrain at Churchill made recovery next to impossible. Over 30 per cent of the area is water during the warm months. And if the rockets don't land in the water, they are likely to land in crevices just as inaccessible to spotting. Or if they do land on the ground, there are so many big rocks and short trees in the area that look like rockets, says Townsend, that you could drop an ICBM in there and never find it. Moreover, much of the land is so soft in the summer that rockets just disappear into the tundra with a

Tundra Test Site

Fort Churchill, site of the recent pre-IGY rocket launching tests, sits on the western edge of Hudson Bay in the province of Manitoba, Canada.

Most of its 500 nonmilitary inhabitants occupy themselves with fishing and gathering furs. A few hunt whales in Hudson Bay. For six weeks in late July and August, when the Bay is open to navigation, Fort Churchill serves as a port principally for the shipment of grain.

Because of its location in the subarctic and in the middle of the aurora zone, Churchill appeared to be a choice site for investigation of the upper atmosphere. Up to this time, upper air research had been confined essentially to desert regions in the southwest United States; the arctic atmosphere and auroral phenomena were practically unknown.

Thus, the National Committee for the International Geophysical Year eagerly seized upon Canada's invitation to use Fort Churchill as an IGY test site. And, starting next July, the base will serve as one of the principal launching areas for high altitude research rockets.

Taxi calls meant night firings.

"squoosh." In the winter months, cold weather and blizzards make it dangerous to send recovery crews out, and the snow quickly covers the rockets.

As a result, says Townsend, there is very little shooting for recovery; most data must be telemetered. Recovery might be feasible if the rockets were equipped with parachutes and electronic locators, in which case other interesting experiments could be run. But, as now planned, most tests will be run without hope of recovery.

The use of telemetering and tracking devices, however, was not without its share of problems too. In addition to the usual cold weather malfunctioning, the systems were plagued by interference. Using a doppler frequency of about 40 megacycles, for example, Fort Churchill technicians found that signals from Patrick AFB (Fla.) and White Sands Proving Grounds (N. Mex.) were mixing with ground-to-rocket signals from their own tracking system.

Even Charlotte (N. C.) taxicabs were cutting in on the Churchill transmissions. Consequently, most firings were rescheduled for nights and weekends when interference was at a minimum.

Count-down: Despite these numerous and vexing problems, scientists at Fort Churchill experienced only one major mishap, were able to get off six of the seven rockets that were scheduled for this preliminary run-through of the IGY. Of the six firings, all were considered eminently successful.

Here, in some detail, is a playback of the recently concluded check-out program at Fort Churchill:

• October 20. An Air Force Nike-Cajun, instrumented for pressure, temperature, and density readings, was fired principally for familiarization and check-out of the whole facility (see picture). Rocket carried approximately a 100-lb payload to an altitude of about 70 miles.

• October 23. An Air Force Aerobee, intermediate model, reached an altitude of about 90 miles with a 200-lb payload. This vehicle was also instrumented for pressure, temperature, and density readings. Although considered a good firing, pressure from the booster blast started to spread the metal sheets covering the launching tower.

• November 5. A full-blown RV-N-13B, Naval Research Laboratory Aerobee-Hi exploded while holding X-minus-15 min, tossed 700 lb of acid around inside the tower, and threw shrapnel through the sides. The rocket carried 220 lb of auroral particle instrumentation. Damage to the launching tower resulting from the explosion (and fire) was repaired.

• November 11. Signal Corps Engineering Laboratory fired an old model Aerobee in a grenade experiment (see picture) to measure temperature and winds. The rocket reached an altitude of about 45 miles with a 220-lb payload. Because this firing was so successful, the Signal Corps called off the other launching they had scheduled. But this firing really blew out the sides of the recently repaired launching tower, made it extremely difficult to maintain temperatures.

• November 15. Naval Research Laboratory group launched another RV-N-13B Aerobee-Hi instrumented for ionosphere exploration. Carrying a 180-lb payload, the rocket reached an altitude of over 80 miles. Although this was considered underperformance, it was adequate for the experiment.

• November 17. Another NRL RV-N-13B Aerobee-Hi, instrumented for pressure, temperature, and density readings, achieved near-perfect performance, reaching 130-mile altitude with 190-lb payload.

• November 20. The last rocket in the test program, a RV-N-13C Aerobee-Hi, actually hit peak performance—160-lb payload to 157-mile altitude. Unlike the other models, this Aerobee-Hi was equipped with proper tankage for its thrust chamber; i.e., there was no fuel left over at the end of its run. The vehicle was instrumented with three mass spectrometers to measure composition of air and ions at high altitudes.

Full significance of these preliminary tests, of course, must await complete analysis of the collected data. But it is already evident that the program fulfilled its primary purpose of equipping members of SCIGY (Special Committee for the International Geophysical Year) with invaluable background for the forthcoming IGY rocket program, a major part of which (over half the solid rocket launchings) will take place at Fort Churchill starting next July.

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...developing missile component and sub-system integration plans and engineering change procedures.

Ground Support Equipment
...responsibility for planning GSE, with emphasis upon problems of compatibility and integration.

Training
...including the preparation of training program criteria and plans, and technical manual specifications.

Operational Planning
...developing missile operational plans and facility criteria.

These areas are a part of Ramo-Wooldridge's systems engineering and technical direction responsibility for the Air Force Intercontinental and Intermediate Range Ballistic Missiles. Engineers with senior project and field experience are invited to explore these positions.

Please address inquiries to: Mr. William Coster

The Ramo-Wooldridge Corporation

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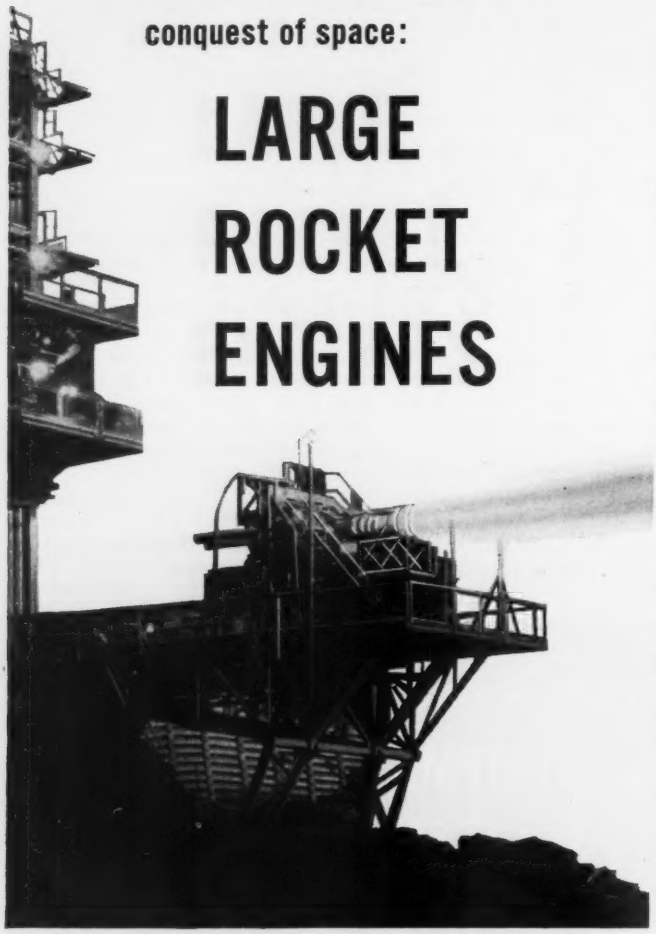
Wanted: Pioneers for man's last frontier

Help us build power for the conquest of space:

LARGE ROCKET ENGINES



Tomorrow's count down already fills the air at ROCKETDYNE's 1,600-acre Field Test Laboratory in the Santa Susana Mountains near Los Angeles. For this is the free world's largest workshop for rocket engineering—the great new industry that is now attracting many of the finest scientific and engineering minds in the country.

EXACTING RESEARCH, EXCITING PROSPECTS

From the rock-bedded test stands come 2 miles of recordings per day—data far ahead of available texts. The big rocket engine is a flying chemical factory in an absolute state of automation. It tolerates no error. It demands ductwork, turbomachinery, pressure chambers, orifices, injectors, heat exchangers and closed-loop control systems that must put hundreds of pounds of precisely mixed propellants into controlled combustion every second. Tolerances go down to 0.0001". Temperatures range from -250° F to 5000° F. Process time constants occur in "steady state conditions" of the order of a few milliseconds. Event sequences are minutely evaluated, as basis of designed performance predictions of extreme exactitude.

The methods now being developed at ROCKETDYNE for producing effective power to the limits of mechanical stress will have wide application. Such experience is practically unobtainable anywhere else. As a graduate engineer, *you* may be able to participate—now.

What motivates a rocket engineer? Well, the material advantages are high; but it is the work itself that draws him most. He feels the same incentive that moved Magellan . . . spurred the Wright Brothers . . . and beckoned again to Goddard as he flew the first liquid rocket at Auburn, Mass. in 1926.

At ROCKETDYNE, you can do this kind of pioneering in a management climate that stimulates personal growth—and rewards it to the limits of your ability. Academically, too, you can grow with our financial aid; some of the nation's finest universities are close by.

INTERESTING BOOKLET: "The Big Challenge"—facts on design criteria and development approaches used at ROCKETDYNE. Write for your personal copy, specifying your degree and years of post-college experience. Address: A. W. Jamieson, Engineering Personnel Dept. **1-JP**, 6633 Canoga Ave., Canoga Park, California.

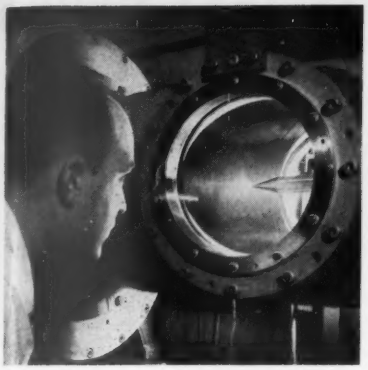
WILLIAM J. CECKA, JR., 35, aeronautical engineer, (Univ. of Minn. '43), was called from North American by the Air Force for experimental rocket work in 1944. On his return, he progressed rapidly: 1948, supervisory test job; 1950, group engineer, operations; 1953 engineering group leader; 1955, section chief of engineering test. Using our refund plan, he has his M.Sc. in sight.

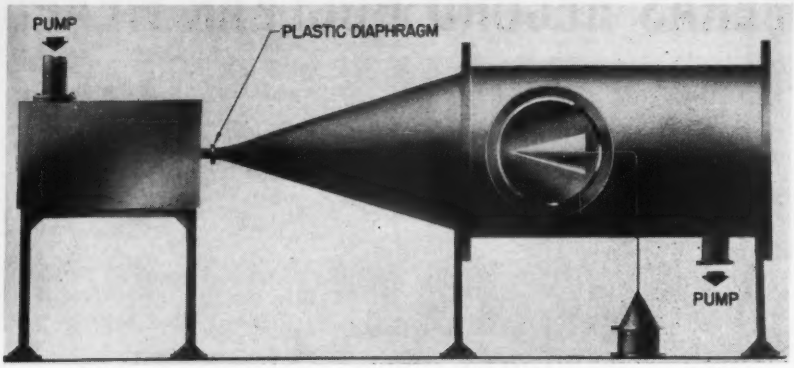
GEORGE P. SUTTON, in the 13 brilliant years since receiving his MSME, Cal Tech, has made rocketry a way of life. His reputation is world wide. His book *Rocket Propulsion Elements* is recognized as the standard text on the subject. Still active academically, but no bookworm, he takes time off occasionally to study the laws of motion at some of the world's better ski resorts.

ROCKETDYNE

A Division of North American Aviation, Inc.

BUILDERS OF POWER FOR OUTER SPACE





"Hot Shot" tunnel simulates 11,000-mph speeds and temperatures to 15,000 F.

Brief View of Hypersonic Research

ROCKETS, having sundered the sound barrier, are now running into the aerodynamic heating problem.

Some of the more acute physical phenomena encountered behind a normal shock wave are listed in the table at the right. More significant, however, is the adjacent drawing (far right) which shows some of the problems that might be met by an ICBM warhead traveling at hypersonic velocities.

One of the biggest technical problems remaining in the building of an ICBM is, of course, finding some way of keeping the warhead from disintegrating under the terrific heat encountered upon reentry. Research indicates that blunt noses are preferable to streamlined ones in beating this heat problem. But this is far from a full answer.

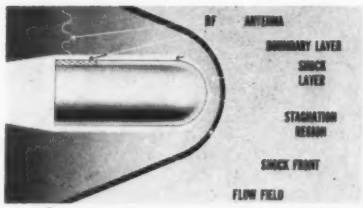
To get a more complete answer, missile scientists have had to devise some rather ingenius wind tunnels which could generate the necessary hypersonic flows. The "Hot Shot" tunnel (above, right), for example, simulates speeds to 11,000 mph, produces temperatures to 15,000 F. In this tunnel, built by ARO, Inc., for Arnold Engineering Development Center, a 1-million amp current shatters a plastic disk in an arc chamber, releasing a 10-millisec flow which blasts into the 16-in. section holding the test model. (Series of photos, below, shows tunnel in action.)

On the other hand, in NACA's new high velocity light-gas gun at Ames Aeronautical Laboratory, the model itself moves (see two photos above). Briefly, this is how it works: First, the model is placed in the barrel; then helium under high pressure "shoots" the model down the barrel—at speeds to 10,000 mph—into a pressurized tank at the far end. A special blast chamber absorbs the outrush of gas from the gun barrel.

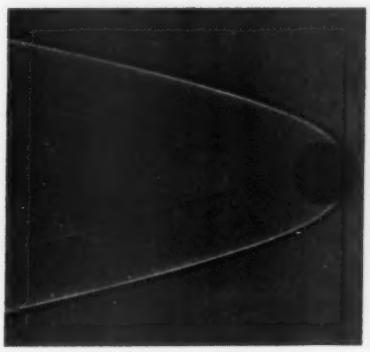
This, of course, is only a small part of the picture. At laboratories around the U. S., engineers and scientists are continuously seeking for better materials and designs with which to push missiles through the thermal thicket and toward, perhaps, another barrier.

MACH NO.	PRESSURE RATIO	TEMPERATURE RATIO	TEMPERATURE °K	DENSITY RATIO
5	29	6	1600	5.5
10	130	16	4400	8.4
15	290	27	7400	9.7
25	800	44	12000	12.1

Some of the conditions prevailing behind a normal shock moving in air.

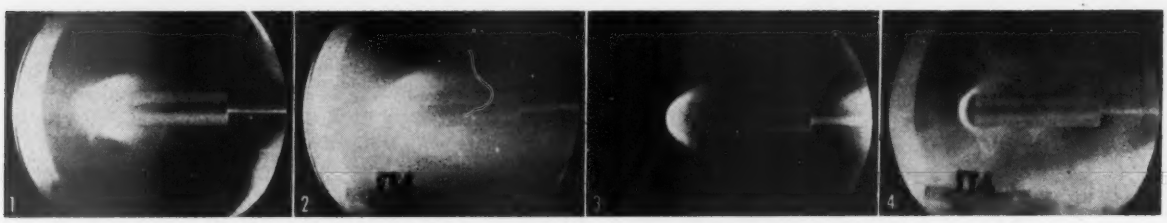


Drawing shows schematically conditions which might be met by ICBM warhead.



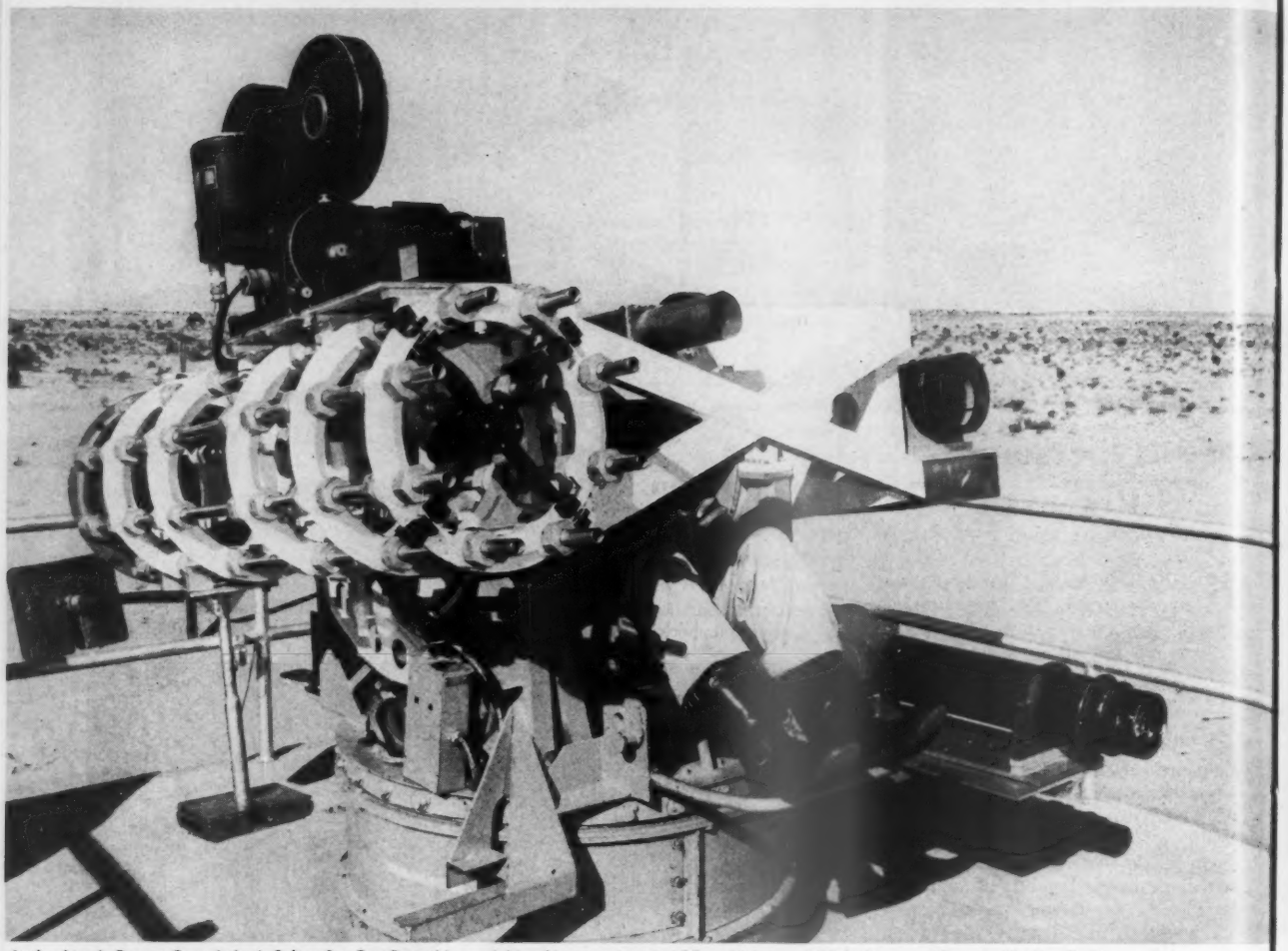
Shadowgraph (at one ten-millionth sec) shows flight of a sphere at 10,000 mph.

Blast chamber in Ames light-gas gun absorbs outrush of gas from gun barrel.

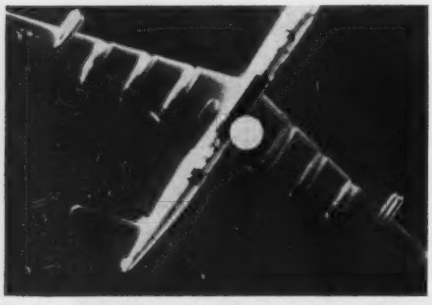


Series of photographs shows distribution and behavior of air flow over a test model at 11,000 mph and 20,000 psia. In sequence, airflow starts over rounded model nose, reaches peak velocity, loses speed and luminosity, ends with nose still glowing hot.

CAMERAS RECORD NUCLEAR WEAPONS TEST DROPS



At the Atomic Energy Commission's Salton Sea Test Base this special tracking mount uses a 35mm high speed Mitchell camera. Operated by Sandia Corporation, this base uses Mitchell cameras for recording test operations.



Tracking at 40,000 feet, this film frame from a Mitchell camera operating at 96 frames per second provides an accurate record.

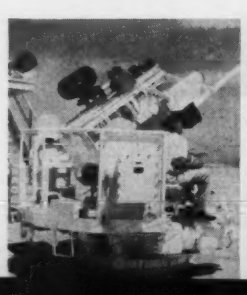
Salton Sea Test Base Uses Mitchell Cameras to Capture High Speed Action of Dummy Bombs

High speed flight and laboratory tests, which hitherto have been difficult or impossible to view with the human eye, are today providing revealing information through high speed film recordings.

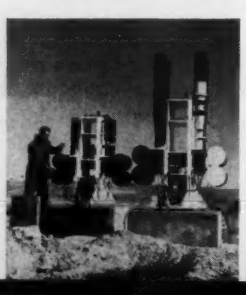
Typical example of the widespread use of high speed cameras is the Salton Sea Test Base in Southern California, where drop testing of "dummy" bombs is a major activity. In testing carried on there, by Sandia Corporation for the Atomic Energy Commission, as many as 20 Mitchell high speed cameras may record different angles in the flight of an experimental weapon "shape" from drop aircraft to impact area.

Operating at 48 to 100 frames per second, the Mitchell cameras film accurate, steady images with maximum uniformity—even under difficult and complicated filming conditions.

Mitchell cameras play a growing role in today's research and development—just as 16mm and 35mm Mitchell cameras have become the leading professional motion picture equipment used by industry, television, and film studios throughout the world. Write on your letterhead for further information on the uses of Mitchell cameras in the field of military and industrial research.



High speed Mitchell Camera in operation on tracking telescope mount during test run at Salton Sea.



Photogrammetric mounts for 8 Mitchell cameras determine the position in space of a nuclear explosion.

Mitchell Camera
CORPORATION
666 WEST HARVARD STREET
GLENDALE 4, CALIFORNIA
Cable Address: "MITCAMCO"

Jet Propulsion News

Alfred J. Zaehringer, American Rocket Company, *Associate Editor*

MISSILES

• Army has renamed its series of Nike antiaircraft missiles. Nike I is now Nike-Ajax; Nike B becomes Nike-Hercules; and Nike II is Nike-Zeus.

The army also disclosed that the first set of Nike-Hercules ground guidance equipment has been released from Western Electric Co.'s Burlington (N. C.) plant and will be installed at existing Nike sites to bolster defense around key American cities. The missile itself is scheduled to go into operational use shortly.

• Flight control system for Lockheed's X-7 ramjet test vehicle (see JET PROPULSION, Sept. 1956, p. 793) comes from Lear, Inc.

• Douglas Aircraft Co.'s new test facility in the Sacramento (Calif.) area will be used for development work on the USAF's Thor IRBM.

• A recent memo signed by Secretary of Defense Charles E. Wilson assigns control of long-range (over 200 miles) missiles to the Air Force. It does not, however, prohibit the Army from "making limited feasibility studies in this area."

• The Army will soon begin construction of Nike facilities at the USAF base in Thule, Greenland.

Meanwhile, firepower at four of the 12 Nike sites surrounding Philadelphia is being doubled.

• Navy has awarded Bath Iron Works contract for construction of two guided missile frigates, DLG 14 and DLG 15. The ships will be armed with Terrier missile batteries aft and five-inch gun batteries forward.

• First step in the development of a fleet ballistic missile was taken last month with the launching of the experimental navigational ship USS Compass Island (EAG 153). A converted Mariner-class cargo vessel, the ship is equipped with advanced navigational and stabilizing gear. Its objective is to assist in the development and evaluation of a navigation system independent of shore-based aids.

• Boeing Airplane Co. has optioned acreage near Parks Air Force Base (Calif.) for possible construction of its Bomarc missile.

• Firing and launching platform for Project Vanguard is now in place. The unit was designed, built, and installed by Loewy-Hydropress Div. of Baldwin-Lima-Hamilton Corp.

• Convair Div. of General Dynamics Corp. will test USAF Atlas ICBM at four facilities now under construction in Southern California (two at firm's San Diego facilities and one at Edwards AFB) and Florida (Patrick AFB). Actual launchings will take place only at Patrick.

• On Dec. 8, the Navy fired a modified Martin Viking from Patrick AFB in a check-out of some equipment that will be used in Project Vanguard. Similar to (but not a prototype of) the first-stage Vanguard rocket, the Viking reportedly reached an altitude of 125 miles.

In last month's test the government also tried out some new launching equipment. It included a curved steel apron over which water flows to divert the rocket's hot exhaust during launching. The test, said the government, went according to plan, and the apron was found to be satisfactory

• In another test from Patrick AFB last month, a Northrop Snark—programmed for a closed-circuit flight over the Atlantic—failed to make a turn and is believed to have ended its flight some 3000 miles away in the jungles of Brazil.

COMPANIES

• Norton Co. officially opened its new electric furnace plant in Huntsville, Ala. Some of the materials to be produced here are boron carbide, fused zirconia, fused magnesium oxide, and fused alumina. Because of their ability to withstand wear at high temperatures, many of these materials will be of interest to missile engineers.

• Minneapolis-Honeywell Regulator Co. has purchased Davies Laboratories, Inc., of Beltsville, Md., manufacturer of specialized high-speed data recording systems used in missile and jet testing.

• Walter Kidde & Co., Inc., has leased a 22-acre track in Clifton, N. J., to provide growing space for the firm's work on development of secondary power units for guided missiles.

• Bendix Aviation Corp. has established a new operating division at Ann Arbor (Mich.) which will concentrate on weapons systems requirements of the Department of Defense.

• Consolidated Diesel Electric Corp. is slated to begin production on a $100,000 hydraulic test stand for Allison Div. of General Motors. The company also has an order for three portable hydraulic test stands to support Martin's Matador missile.

• Weber Aircraft Corp. has received an additional order of more than $1 million for upward-ejection seats for the Boeing B-52.

• A new company, Propellex Chemical Corp., has been incorporated in Edwardsville, Ill. The firm will concentrate on research, development, and production of propellants and explosives.

• Aerojet-General Corp. has changed the name of its Electronics and Guidance Division to Avionics Division.

In another change, the firm promoted former ARS Director John B. Cowen from Director of Government Operations to Assistant to the Vice-President, Manufacturing.

• Lockheed Missile Systems Div. has occupied another new plant. This one is located at Sunnyvale, Calif.

INSTITUTIONS

• In a joint effort with International Business Machines Corp., New York University will hold a two-day (Jan. 31 and Feb. 1) symposium on digital computing in the aircraft industry.

• The 1957 Nuclear Congress will be held March 11–15 in Philadelphia's Convention Hall.

• Armour Research Foundation is investigating sound propagation through the atmosphere and the effects of various weather conditions upon this propagation.

To facilitate another investigation (of rocket and jet engines and fuels), Armour has set up a new combustion laboratory.

• University of California is planning to hold its Ninth Annual Industrial Engineering Institute on Feb. 1 and 2 at its Berkeley and Los Angeles campuses. Purpose of the sessions is to bring industrial engineers and managers up to date on the latest developments in research and practice in their fields.

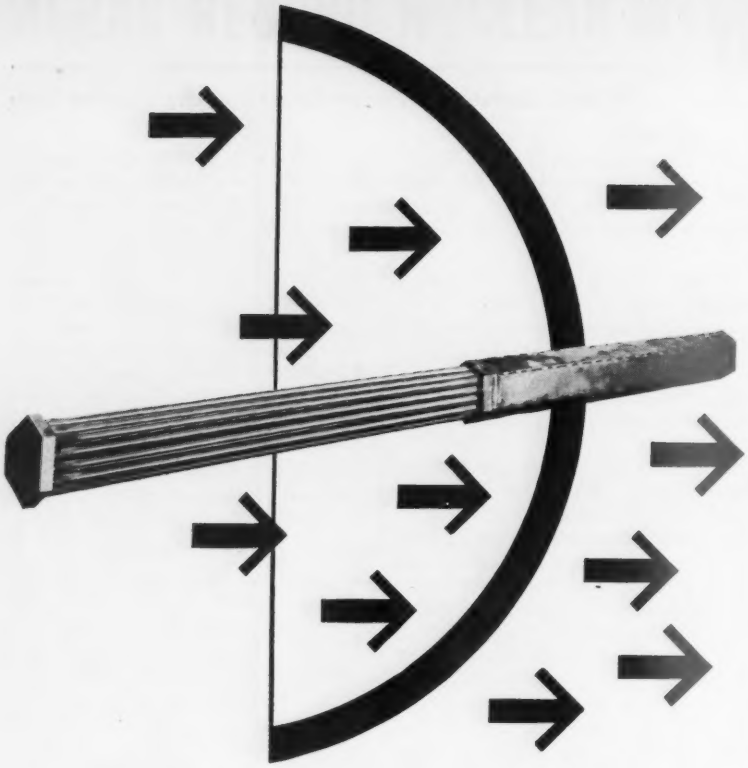
• Seven of the twelve-man board of the American Astronautical Society have resigned. Among those leaving the group are Wernher von Braun, Krafft A. Ehricke, and S. Fred Singer.

• William F. Ryan, Stone & Webster Engineering Corp., is the new president of ASME.

• Application forms for Guggenheim fellowships for graduate study in the fields of jet propulsion and flight structures are now available from presidents and deans of all engineering and technical colleges. The Daniel and Florence Guggenheim Foundation has announced that 18 to 20 fellowships will be awarded in 1957.

FOREIGN

The most noticeable feature of the International Congress on Rockets and Guided Missiles, held at the École



want to pioneer?

Fuel element for proposed airborne packaged power unit is typical component in Martin development of transportable nuclear power systems... Program points way to opening of vast new frontiers, with immense future for "packaged power"... Also, Martin nuclear activities now rapidly expanding as result of a major development contract in field of aircraft nuclear propulsion... Many excellent opportunities for engineers with or without previous nuclear experience ... Contact J. M. Hollyday, Dept. JP-1, The Martin Company, Baltimore 3, Maryland.

MARTIN
BALTIMORE

Nationale Supéieure d l'Aéronautique in Paris Dec. 3–8, was the scarcity of United States representatives. The only one much in evidence was A. W. Robinson of General Electric Co. who deputized for Andrew Haley, listed as one of the three presidents of the congress, who cabled regretting indisposition.

There were more than 500 delegates from 19 countries in the list of delegates published half-way through the congress, including 16 from the United States, 22 from Britain, 50 from Germany, 7 from Poland, 4 from Turkey, 18 from Switzerland, and 9 from Russia. Many of those listed had not been seen in the Congress Hall. The organizers blamed the international situation for nonarrivals. Local politics undoubtedly also had something to do with it. Nevertheless, lectures were well attended although, as usual, many had come for what they hoped to learn in the lobbies or over the dinner table. As the conference was at least the fifth in Europe to be devoted to the subject of jet propulsion in the last ten months it was not surprising that little if any new information came out.

The Russians turned up in force and listened with interest while a Swiss delegate described Oerlikon guided weapons systems and an Italian described sodium potassium and boron hydride motors that he thought would find important places in weapons of the future, especially torpedoes. (Russians were due to talk Dec. 6—too late for this issue of JET PROPULSION.)

RESEARCH & DEVELOPMENT

• A new ceramic material, developed by Alfred University and fabricated into nozzle inserts by Electro Refractories & Abrasives Corp., is said to show promise in jet exhaust nozzle tests, may find applications in flameholders as well.

• Hughes Aircraft Co. will build a $1^1/_4$-million engineering facility at its Tucson (Ariz.) missile plant.

• ACF Industries, Inc., will design and fabricate a nuclear reactor that will be used to test irradiation effects on aircraft parts under simulated high altitude conditions at Wright-Patterson AFB.

• Raytheon Manufacturing Co. has established a new electronics laboratory at Maynard (Mass.) which will have as its primary responsibility the design and development of airborne electronics equipment.

• The National Bureau of Standards now has what has been called "the quietiest room in Washington." It is an anechoic chamber built for the Bureau by Eckel Corp. This special sound-absorbing room is designed for studying sound.

NEW DEPARTURE IS A HELPING HAND IN MINIATURIZATION

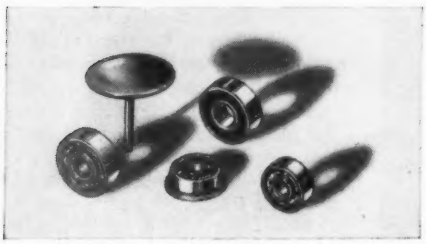
Volume is one of the big reasons why New Departure is the preferred source for miniature ball bearings.

And, of course, another big reason is ultra-precision! Despite their small size ($\frac{5}{32}$" to $\frac{1}{2}$" O.D.), New Departure miniature bearings are fully precision-ground, lapped, and honed to within tolerances of ABEC 5 or better.

Actually, precision starts with proper preparation of the stainless steel, where optimum metallurgical characteristics are assured by exacting heat treat equipment and processes. Finished bearings undergo rigorous testing and microscopic inspection on special equipment. Assembly and packaging take place in completely air-conditioned and pressurized areas to keep out contaminants.

However exacting your performance requirements, call on New Departure for miniature bearings unexcelled for precision, accuracy, and long life. New Departure, Division of General Motors, Bristol, Connecticut.

BALL BEARINGS MAKE GOOD PRODUCTS BETTER



SEE NEW DEPARTURE ON
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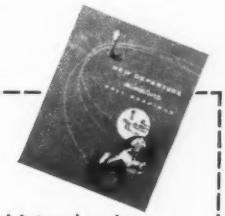

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Seventh IAF Congress Stresses Cooperation

ANDREW G. HALEY[1]

THE Seventh Annual Congress[2] of the International Astronautical Federation, held in Rome, Italy, Sept. 17–22, 1956, was remarkable in many ways. But the organizational effort was outstanding, including excellent conference facilities, efficient use of simultaneous translation (into four languages), and a beneficial camaraderie in the Secretariat.

The chief failure of this Congress was imperfect screening of the papers read to the delegates. While most papers were of extraordinary merit, the rest were weak and not worthy of presentation. The fault, however, may not be attributed to the management of the Congress—undertaken by G. Arturo Crocco, Antonio Eula, and Glauco Partel—which had no real control over the papers submitted by societies of the 21 participating nations. Well-deserved criticism may be directed at the societies themselves.

Steps were taken to correct this serious problem and it is hoped that future screening will be more effective.

Over 450 experts in the natural and social sciences attended the meetings. Approximately 50 papers were presented. The authors included, among others: S. F. Singer (U. S.), J. J. King (U. S.), W. B. Klemperer (U. S.), R. M. L. Baker (U. S.), M. Dubin (U. S.), F. Romano (Italy), A. C. Robotti (Italy), B. Bergqvist (Sweden), J. De Nike (U. S.), N. E. Felt (U. S.), A. A. Cocca (Argentina), A. G. Haley (U. S.), J. Kooy (Holland), N. V. Petersen (U. S.), R. W. Buchheim (U. S.), A. Boni (Italy), K. R. Stehling (U. S.), R. Foster (U. S.), H. J. Kaeppeler (Germany), G. Baumann (Germany), B. Langenecker (Austria), E. De Zubay (U. S.), H. Bednarczyk (Austria), W. Peschka (Austria), H. Ruppe (Germany), K. Ehricke (U. S.), K. Zarankiewicz (Poland), H. Krause (Germany), H. Strughold (U. S.), S. J. Gerathewohl (U. S.), D. G. Simons (U. S.), E. Sänger (Germany), G. A. Crocco (Italy), J. Pressman (U. S.), H. Grosch (U. S.), M. Vertregt (Holland), R. Engel (Egypt), R. M. Corelli (Italy), T. Nonweiler (England), H. E. Hinteregger (U. S.), P. F. Bedinger (U. S.), and Darrell Romick (U. S.).

Three new national societies were elected to membership; namely, Committee on Astronautics, Academy of Sciences (Russia), Polskie Towarzystwo Astronautyczne (Poland), and Société Francaise d'Astronautique (France), making a total of 21 voting members.

The most felicitous occasion of the Congress was the special audience with Pope Pius XII who showed a far-reaching knowledge of the history and activities of IAF and other leading rocket and interplanetary societies.

In reviewing the history and contributions of the IAF, the Pope noted particularly the leading role played by the Federation in bringing about the proposed launching of the artificial earth satellite. He pointed out that God did not intend to limit man's knowledge to the earth alone. Rather He offers to the human mind the whole creation so that man may see through it and "thus may understand always more profoundly the infinite grandeur of his Creator."

But, the Pope added:

"Without delving into details, I might say, Gentlemen, that plans of such a range entail intellectual and moral aspects which cannot be ignored; they require a certain conception of the world, its meaning, its end. . . The most advanced explorations into space will only serve to bring a new reason for disunion if they are not effected along with a deeper moral intention and a more conscious attitude of devotion in the higher interests of mankind."

In conclusion, He blessed the delegates, their families, and associates.

Getting Down to Business: Administratively, the most important achievement of the Congress was the unanimous adoption of amendments to the IAF Constitution. In brief, these amendments increase the authorized number of vice-presidents of the Federation from three to five, and establish new procedures for nominating and electing IAF officers.

For this year the Congress chose L. R. Shepherd as president and Frederick C. Durant III, Teofilo Tabanera, Paul J. Bergeron, A. Sedov, and Julio Marial as vice-presidents.

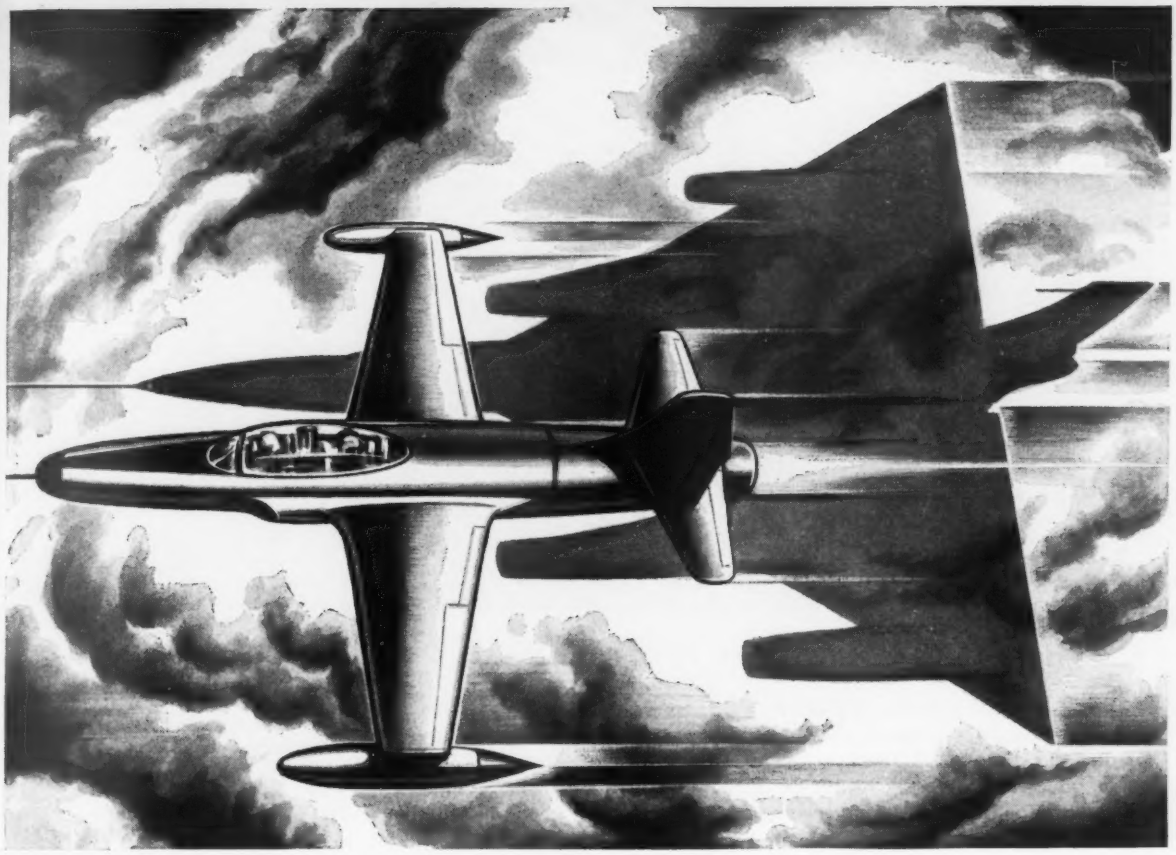
Article 25 has been amended to require that officers shall be members of the voting societies; that no officer other than the secretary may succeed himself except by unanimous vote, and then for one year only; that officers shall be nominated by a committee elected by the Council in plenary session, and consisting of five members of the Council; that no member society shall succeed itself on the nomination committee; that officers shall be elected by secret ballot during the course of the plenary session; that nominations and elections shall be held at each plenary meeting of the IAF Council but not more frequently than once a year; and that the officers shall hold office until their successors are duly elected and qualify.

With respect to cooperation with UNESCO, the IAF worked out a formal agreement whereby it will act as a consultative international nongovernmental organization to UNESCO upon the planning, organization, and exe-

[1] ARS Director and General Counsel; Chairman, IAF International Affairs Committee.

[2] The history of the IAF from the First Congress in 1950 through the Sixth Congress in 1955 was reported in JET PROPULSION, Nov., 1955, p. 627.

God did not intend to limit man's knowledge.



At Cornell Aeronautical Laboratory, The F-94

CASTS A SUPERSONIC SHADOW

Months before the Convair B-58 supersonic bomber took to the air, its handling characteristics had been successfully simulated by C.A.L. on a totally different airplane.

Our study of the predicted performance of the new bomber's proposed control system and configuration was accomplished with a modified F-94 fighter carrying highly-specialized hydraulic, mechanical and electronic components. The stick forces and motions of the B-58, as well as its stability characteristics, were reproduced by this automatic control system, one of the products of C. A. L.'s continuing long-term effort in artificial stability and control research.

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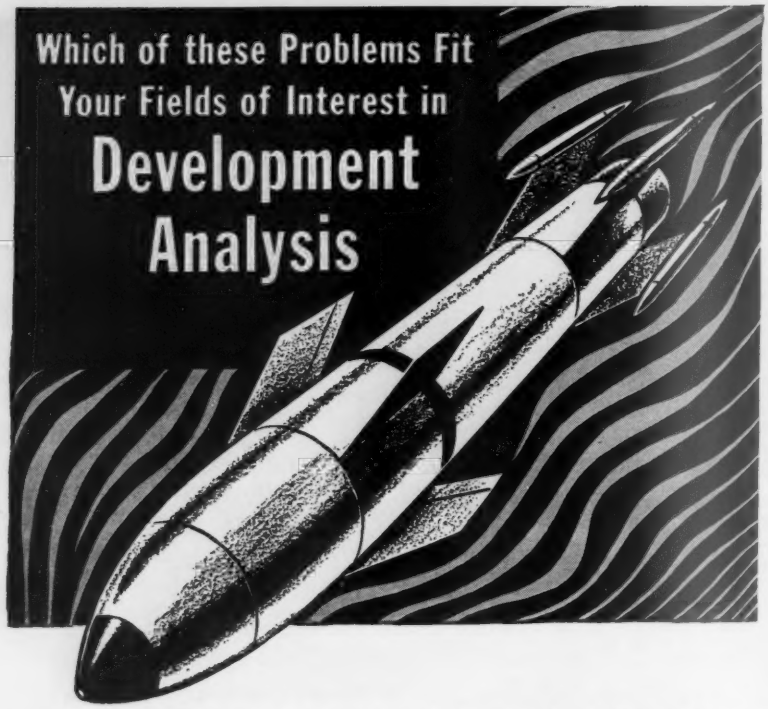

The increased demands placed on airplanes and missiles call for a specific knowledge of aerodynamic and structural behavior at the outset of design activities. The ability to emerge with the aerodynamic design while sucessfully solving problems in performance, air loads, stability, control, maneuverability and aerodynamic heating has been repeatedly demonstrated in the past.

As in Aerodynamics, the increased demands placed on airplanes and missiles require careful consideration and early solutions of structural problems. This involves designing and testing structural components and assemblies, and determining structural design loads, methods and allowables.

Some of the problems which are under extensive investigation at present are:

AERODYNAMICS

1. Fluid flow and boundary layer analysis at hypervelocities and low Reynolds Numbers.

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cution of a program relating to space flight and astronautics in general.

This program was presented to UNESCO by Andrew Haley who was elected official IAF representative to, and who attended, the New Delhi Conference of UNESCO held last November.

An International Affair: In regard to cooperation with other international organizations in the field of astronautics, the IAF wrote to Marco Andrada, secretary general of the International Telecommunication Union, last April, requesting that adequate working arrangements be made between the IAF and the ITU for allocation of radio frequencies for space communications.

A report should be available for the Plenipotentiary Conference of the ITU in 1959. Once such studies are made, the ITU should initiate formal steps to effect the allocation of radio frequencies for travel in space.

Radio communications and guidance, of course, are essential in space flight. In the case of the earth satellite programs it is also obvious that such programs must be peaceful and civilian, as the satellites will orbit over many nations. The only official international body authorized to make studies on the fundamental and extremely diverse technical problems involved is the CCIR.

In view of the importance of the CCIR to the astronautics program, Andrew Haley traveled to Warsaw to attend the CCIR conference last August to present the views of the IAF and to seek associate membership on the CCIR. Dr. van der Pol, the CCIR director, is now circulating the membership on these propositions.

At its convention in Caracas last June, the International Civil Aviation Organization showed real awareness of its possible role in the field of astronautics, recognized the need for the study of space law by that body. P. K. Roy, director of ICAO's Legal Bureau, expressed his interest in a seminar on space law and allied problems sponsored by ICAO.

Throughout 1956 there were many other instances of cooperation and interest in astronautics by international organizations and their members. In January, for example, C. Wilfred Jenks of the International Labour Office published one of the truly great papers on the subject of space law. In December, the National Congress of Electronics, Telecommunications and Broadcasting, convening in Mexico City, considered space travel problems.

And during the past year many other examples of the growing interest and cooperation in astronautics were evinced.

International preparation for the forthcoming International Geophysical Year is the subject of daily news reports.

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Early Work and Latest Realizations With Ramjet Engines

RENÉ LEDUC[1]

René Leduc et Fils, Argenteuil, France

THE attainment of supersonic flight has been, at least up to the present time, a problem of propulsion. Man could fly only when he was able to build light engines of sufficient power.

Developments in aerodynamics have, of course, been helpful; it is noteworthy, however, that progresses made in this direction have contributed more to augment speed than to increase payload.

The fact is that we are dominated by speed. It has cost mankind a great part of its revenue, and no leveling-off period can be predicted. Under these circumstances, it may be wise to have a close look at the problem lest we may be heading into insurmountable difficulties.

That is why, before discussing propulsion systems, I would like to examine the problem of high speed flight. We will see that it is dominated by weight.

Airframes are becoming heavier in order to obtain the required rigidity, and the use of very thin airfoils (5 per cent) does not help the situation. While 20 years or so ago the airframe weight was about 25 per cent of the total weight at takeoff, today we are satisfied when, in a fast airplane, it does not exceed 40 per cent.

The weight distribution is about as follows:

Airframe	40%
Equipment, crew, payload	10
Propulsion system plus fuel	50
	100%

It is apparent, therefore, that under the best conditions, only half the total weight is available for the propulsion system and the fuel.

Let us see now what we need: All airplanes would like to be supersonic—or perhaps we should say that airplanes can be supersonic to a degree. We all agree that every modern airplane must break the sound barrier; of course, one can dive a little if the propulsion system has not exactly got what it takes. There is no harm in it, after all. And why give the preference to the horizontal line?

Before appraising its needs, let us first define the supersonic airplane. It is apparent that, in the near future, we will fly at Mach 2 without too much difficulty. The airplane must then be able to climb at supersonic speed, make 3 g turns at 60,000 ft without losing altitude (this latter requirement is very important for interceptors), and burn fuel at a rate commensurate with flight duration requirements. The drastic change in lift drag ratio from subsonic to supersonic speeds is illustrated in Fig. 1, with

n = g factor
F_n = net thrust of the propulsion system
W_g = weight of the airplane (gross)
$F_n = n \cdot W/(L/D)$

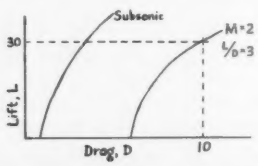

Fig. 1 Change in lift drag ratio from subsonic to supersonic speeds

At $M = 2$ it is difficult to obtain an L/D above 3 and during a 3 g turn $F_n = W_g$.

This relationship must be satisfied at any altitude where the aircraft is expected to effect 3 g turns without losing altitude. Assuming that the engine thrust varies approximately with atmospheric pressure, the thrust required at sea level for the same Mach number is

$$F_{n,o} = F_n(P_0/P)$$

P = ambient pressure
P_0 = sea level atmospheric pressure

It follows that

$$F_{n,o} = n \cdot W_g \frac{P_0}{P} \frac{1}{L/D}$$

If the operational altitude of the airplane under consideration is 56,000 ft (17,000 m) where $P/P_0 = 0.09$, the sea level equivalent thrust equals

$$F_{n,o} = 3 \times \frac{1}{0.09} \times \frac{1}{3} \times W_g = 11\ W$$

Thus, an airplane of 13,000 lb requires an engine capable of 143,000-lb thrust at sea level Mach 2, more than ten times its own weight. We dispose of 6500 lb for the engine plus the fuel!

These simple relationships present the propulsion problem of high altitude supersonic flight in its true light.

Looking at it another way, the sea level power equivalent is expressed by

$$HP_o = \frac{F_{n,o} \cdot V_0}{550} = \frac{11W\ V_0}{550}$$

and per pound

$$HP_0/W_g = 11 \cdot V_0/550$$

at M = 2

$$V_0 = 2300 \text{ fps } (700 \text{ m/sec})$$

$$HP_o/W_g = \frac{11 \times 2300}{550} = 46 \text{ hp/lb}$$

So the question of weight, the nightmare of the airplane builder, is with us today as it was in the early days of aviation; in fact, one thousand times more so, since it is not so long ago that the same factor was only 0.045 hp/lb.

It is evident that nothing would have been possible without an evolution of propulsion systems, but how slow this evolution has been! And why was it so? Primarily, because of the defects or imperfections of any new machine and because man, by temperament or due to laziness of the mind, is essentially conservative; he prefers to use what is available and works satisfactorily rather than try something new. Whether man is right or wrong is a question which is entirely relative. He is right when others are not making any progress; he is wrong in the opposite case.

Let us now consider the case of propulsion by reaction and, more particularly, the case of the turbojet engine. It is safe to say that the future failings of the reciprocating engine-propeller combination could have been forecast long ago. We warned against them in 1934 in a bulletin of the Academy of Sciences; yet it took World War II to establish the superiority of the turbojet engine! War is a serious business during which there is no time for academic considerations. Today, faced by the fatidic 50 per cent weight allowance and by the enormous power requirements of high speed flight, the airplane builder does not quite know what patron saint to

A lecture presented at the ARS Semi-Annual Meeting, Cleveland, Ohio, June 18–20, 1956.
[1] Engineer.

turn to, and he installs propulsion systems as light as possible without regard for thermal efficiency. He happens to be right, for speed is the primary objective. This is how the turbojet was born. This engine, which was long awaited, has rapidly attained an extraordinary stage of development and it is perfectly adequate for current sonic airplanes. Are things going to stay that way? Certainly not. Truly supersonic airplanes are a military necessity, and man would not be what he is if having the possibility to go faster he would neglect to do so. The equivalents of thrust and power, previously mentioned, appear awesome at first; we will see that they can be obtained with ramjet engines. Moreover, this affirmation is not only based on calculations but on the results of tests conducted in our supersonic wind tunnel.

Ramjet engines constitute a subject in which I'm particularly interested although I have also built small gas turbines (a turbine of 200 hp was running in 1940). But before delving into it, I would like to retrace the development of our ideas on propulsion by reaction.

Soon after our contact with the aviation field in 1924, and after conducting some work in the field of elasticity, we endeavored to find a propulsion system to replace the reciprocating engine-propeller combination. Unfortunately, the piston engine was already at that time too perfect a thermic machine. In addition, no one supposed at that time that weight might some day impose a limit on speed, and had we known what this limit was, nobody would have been disturbed. Our minds were further blindfolded by the concept of autonomy, of which powered takeoff was a necessary part. As for me, I do not regret it, for it led me in 1929 to the study of a pulsejet engine capable of interesting static performance. At that time, I wrote a memorandum in which it was shown that an over-all efficiency of 10.5 per cent was possible at 450 mph and I proposed the use of this type of engine to increase the speed of racing airplanes from 340 to 450 mph. It was the era of the Schneider Cup. It would be superfluous to go into why this system was judged without interest at the time. Nonetheless, I took a patent on it in February 1930, a few months before Paul Schmidt, the inventor of the V-1 powerplant, applied for his patent on April 24 of the same year. Of course, in such machines, the idea is nothing, and Paul Schmidt had the merit to develop it, which was probably not very easy!

Still blindfolded by the need for autonomy and, hence, powered takeoff, I had the idea in 1932 to use a nozzle; namely, a simple tube having a hot and a cold source. It can be shown that such a nozzle can produce a very large thrust under static conditions and without moving parts. I investigated the idea in further detail and on June 7, 1933, applied for a patent describing a principle for the transformation of thermal energy into kinetic energy by means of a hot and cold source. Soon after, I realized that in the practicable temperature range, it was almost impossible to remove a useful quantity of heat from a fluid in motion without prohibitive losses. However, it was possible in a flying machine to realize a cycle releasing energy without a cold sink, and I continued my studies on this type of engine. Prospects looked so promising that I decided to work full time on this particular problem. I advised Professor Villey of my work and findings. At the Sorbonne, Professor Villey had been my counselor and examiner of my thesis for Ph.D.; we discussed in great detail all these questions which later became the object of a paper presented before the Academy of Sciences. At that time, we did not know that another Frenchman, René Lorin, had a similar idea and had published in *Aerophile* in 1913 a short article on this question. René Lorin had a very inquisitive mind and held prophetic views of the future. To honor the memory of these two scientists, my first ramjet airplane, Model 010, which is now at the Air Museum in Paris, bears the name "Jean Villey et René Lorin."

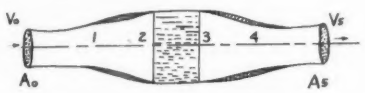

Fig. 2 Ramjet engine

The ramjet does not differ very much from other engines; in order to release work, it needs the customary phases of compression, heat input, and expansion. Its only peculiarity is the absence of moving parts, the compression being due solely to the speed of translation; hence, the name ramjet.

The operation is as follows: The air enters the system at a velocity V_0 and is brought through a diffuser to a lesser velocity V_1 which results in an increase in both static pressure and temperature. Heat is added by combustion raising temperature from T_2 to T_3, and the burned gases are then allowed to expend back to ambient pressure through the exhaust nozzle. The net thrust obtained is equal to the air mass flow times the change in momentum

$$F_n = \frac{W_a}{g}(V_5 - V_0)$$

F_n = net thrust
W_a = air flow per second
ρ = density, unit weight per unit volume
A = passage area
M = Mach number

For continuity and neglecting the weight of fuel

$$V_0 \cdot A_0 \cdot \rho_0 = V_5 \cdot A_5 \cdot \rho_5$$

therefore

$$F_n = \frac{V_5^2 \cdot \rho_5 \cdot A_5}{g} - \frac{V_0^2 \cdot \rho_0 \cdot A_0}{g}$$

let

$$X = \frac{V_5^2 \cdot \rho_5}{V_0^2 \rho_0} = \frac{M_5^2}{M_0^2}$$

since

$$P_{s,0} = P_{s,5}$$

we have

$$F_n = \frac{V_0^2 \cdot \rho_0}{2g} 2(X A_5 - A_0)$$

Now if we call $C_{x,T}$ the thrust coefficient of a ramjet engine propelling an airframe of surface S we have

$$F_n = C_{x,T} \cdot S \cdot V_0^2 \cdot \rho_0 / 2g$$

so that

$$C_{x,T} = \frac{2}{S} \cdot (X A_5 - A_0)$$

The thrust coefficient we have just defined has a remarkable characteristic in that the thermal process is not involved in it (at least directly). One can say that the thrust of a ramjet is due to the fact that, the flow being consistent with continuity, the exit area is larger than entrance area. One can further appraise the fact that, looking at the foregoing sketch, the points of application of the pressure forces are on the inner diffuser wall.

Naturally, it is not possible in flight to obtain a flow consistent with continuity with an exit area larger than the inlet without bringing combustion (or heat transfer) into play; in other words, the production of energy requires a source of heat. However, the problem is different in a wind tunnel and the remarkable expression C_xT above has made it possible for us to discover a method of testing models in a wind tunnel without combustion. This method is of particular interest for high Mach number evaluation where the size of models becomes so small that it precludes the use of combustion.

It can be shown that

$$\frac{A_0}{A_5} = \sqrt{\frac{\zeta X}{\theta + 1}}$$

where

$$\zeta = \frac{1 + \left(\frac{K-1}{2} X M_0^2\right)}{1 + \left(\frac{K-1}{2}\right) M_0^2}$$

$$\theta = \frac{T_3 - T_2}{T_2} \quad \text{(relative heating)}$$

The value of ζ is generally close to unit at least for small Mach numbers.

The thrust coefficient $C_{x,T}$ can be expressed

$$C_{x,T} = \frac{^2A_5}{S}\left(X - \frac{A_0}{A_5}\right) = \frac{^2A_5}{S}\left(X - \sqrt{\frac{\zeta X}{\theta + 1}}\right)$$

This formula illustrates the influence of the thermal process. Noteworthy is the fact that for a given value of θ, the thrust coefficient is constant. This says in effect that the thrust of a ramjet engine is proportional to the velocity squared. Thus, the ramjet is always able to insure propulsion of the airframe, whatever the speed and altitude may be, and we are not any more limited by the weight of the powerplant.

For $V_0 = 0$ the thrust is zero and auxiliary means are required for takeoff. This is the ransom of high speed.

I'd like to emphasize again that together with the rocket engine the ramjet is the only powerplant that can insure flight whatever the order of magnitude of the speed.

The over-all efficiency of the thermal propulsion system is the product of the thermal efficiency and the propulsion efficiency

$$\eta_0 = \eta_t \cdot \eta_p$$

It can also be written

$$\eta_0 = \frac{\xi V_0^2}{gJC_pT_2}$$

If C_p is constant during compression and expansion

$$\eta_0 = 2\xi \frac{0.2M_0^2}{1 + 0.2M_0^2}$$

where

$$\xi = \frac{\sqrt{(X/\zeta)(\theta + 1) - 1}}{\theta}$$

This term depends primarily on the losses through the ramjet engine (through X and ζ) and to a lesser degree on the relative heating θ.

Let W_f be the fuel flow in lb/hr, H_c the heating value of the fuel, J the mechanical equivalent of heat; it follows that

$$\frac{W_f}{3600} H_c J \eta_0 - F_n V_0$$

and the sfc

$$\frac{W_f}{F_n} = \frac{3600 V_0}{H_c J \eta_0}$$

or

$$\text{sfc} = \frac{3600\, gC_pT_2}{H_c \xi V_0}$$

Assuming a value of $H_c = 18{,}000$ Btu/lb

$$\text{sfc} = 6.45 \frac{C_pT_2}{\xi V_0}$$

Preliminary Tests

We started testing ramjets as early as 1934.

Our superb ignorance of the subject turned out to be a blessing; for it is probable that had we known that flame velocity was of the order of a few feet per second and that a certain author, none other than Stodola, had written that the compression of a fluid by slowing it down is always done with poor efficiency, we would not have dared to go ahead with the idea.

After numerous difficulties, we obtained in 1935 what looked like the correct answer on thrust, but combustion was still occurring in pulsatory manner.

In 1936, we achieved smooth combustion. The values of thrust and consummation obtained were those predicted.

Our experimental ramjet was 1.2 in. in diam at the inlet and 1.52 in. at the outlet.

We obtained the following results under a simulated Mach number of 0.965

$F_n = 8.35$ lb
sfc = 2.85 lb/hr/lb

The test was conducted under optimum conditions where $\theta = 1.2$; also $X = M_5^2/M_0^2 = 0.865$.

The over-all efficiency was $\eta_0 = 0.094$.

In 1934 these results which are still honorable today looked catastrophic to some people. Few realized that the attainment of high speed required extremely light propulsion systems.

The 010 Airplane

At the start, the Air Ministry (Services Techniques de l'Aéronautique) recognized the importance of our experiments. It is my pleasure and duty to express my gratitude toward them publicly. They deserve credit for supporting our ideas when prominent members of great institutions did not think that the reciprocating engine could ever be dethroned, although at the same time Whittle was beginning his work which was to give Britain the position of eminence that we all know.

Confident in the results of our laboratory experiments, we proposed the construction of the 010 airplane in 1937.

In 1940, the ramjet engine, namely, the diffuser, the combustion chamber and the turbine to drive accessories, and the accessories themselves, was completed. Unfortunately, we had to make plans to leave Paris and move south (for reason well recorded by history). After many mishaps we landed near Toulouse in the south of France. There, the material was hidden in barns and my colleagues and I waited for better times to come. Once the war was over, we had to start all over again as a part of the material had been destroyed during bombings.

At last, on April 21, 1949, with Jean Gonord at the controls, the 010 made its first power flight and demonstrated the remarkable possibilities of the ramjet engine. It was the greatest moment of my career. It was the first time that a man was mastering an airplane propelled solely by fire. I felt humble indeed aboard the chase plane which was unable to follow. I was responsible for the life of a man who had put his trust in me and in my machine.

In 1952, the 010, which was then 15 years old, could not be followed by contemporary fighters. With it, we exceeded Mach 0.84 and we could maintain this Mach number with a rate of climb of 8000 fpm at 26,000 ft. Our predictions had been realized

$$\text{sfc} = 3.24 \text{ at } M_0 = 0.84$$

The 010 weighs 6600 lb and has a wing area of 173 sq ft.

The building of a conventional airplane does not present any major difficulties. However, with an airplane as new as this one was we had to design and build all accessories. The large fuel flows to be handled were completely out of proportion with current pre-war practice. It was, therefore, necessary to design and build the entire fuel system including controls. Power was required to drive the fuel pump and accessories. Only a gas turbine was capable of meeting all requirements of space, power, and weight, but it was unknown in aeronautical practice in 1937. We started work immediately on what was probably one of the first turbines built for aircraft application. It was running in 1940. We were also forced to design and build the landing gear.

It appeared to us that control boosters would be necessary to meet high speed aerodynamic loads. We came out with a successful design which is being further developed in collaboration with the Jacottet Company. This type of servo-booster is used today on most French prototypes and on some foreign airplanes.

Operation at high altitudes requires that the pilot be housed in an airtight, air-conditioned cabin which can be jettisoned and parachuted to safety. All these problems had to be, and were, resolved before the 010 could fly as planned. As previously mentioned, this prototype is now at the Air Museum in Paris.

The 021 Airplane

The experimental 021 airplane is now undergoing flight tests at our test center near Istres, not too far from Marseilles. Its wing area is 238 sq ft and it weighs 13,200 lb. Like the 010, it is carried "piggy-back" by a launched airplane.

This solution makes for lightness of the structure. The 021 was shown at the presentation of French prototypes at the 1955 Salon de l'Aéronautique in Paris.

On the 010, everything had been sacrificed for aerodynamic performance, and as a result visibility was greatly reduced. It took exceptional test pilots, like the one I was fortunate enough to have, to climb on board this machine.

On the 021, visibility has been much improved. We have not yet exceeded Mach 0.9, but the climb speed near the ground is close to 40,000 fpm (200 meters/sec). With ramjets, the rate of climb is always considerable; so is the level speed.

The 021 constitutes the last subsonic airplane I will build. It was not possible for me to go from the 010 to a truly supersonic airplane without this intermediate step.

As you can imagine, many problems exist. The pumping of fuel, for instance, which at first looks simple, gives us some worries at times.

Comparison of Propulsion Systems

When comparing propulsion systems, one has to consider their range possibilities and, whatever the system

$$R = H_c \, J \eta_0 \frac{L}{D} \log_e \left(\frac{1}{1 - W_f/W_g} \right)$$

or

$$R = \frac{3600 \, V_0}{\text{sfc}} \frac{L}{D} \log_e \left(\frac{1}{1 - W_f/W_g} \right)$$

where

R = range
H_c = heating value of the fuel
J = mechanical equivalent of heat (778 ft-lb)
η_0 = over-all efficiency
L/D = lift-drag ratio
W_f = weight of fuel
W_g = airplane gross weight
sfc = specific fuel consumption
V_0 = airplane speed

It can be seen from these formulas that the best system is not necessarily the one having the lower sfc; the relative weight of fuel is of primary importance.

But we must remember that

$$\frac{W_E + W_f}{W_g} = 0.5$$

W_E = engine weight

This situation is so overwhelming that it has been necessary to rejuvenate the turbojet engine, which has just been born, by grafting to it a ramjet—they call that post-combustion because, do or die, we had to gain about 100 mph in the supersonic range without disturbing too much the ratio W_E/W_g. At that time, the sfc increased to the level of 2.0. One can already sense that the turbojet is nearing the maximum speed limited by its own weight. Its shortcoming is that its thrust does not augment with the speed squared. All this proves that the specific fuel consumption is not as important as some people thought and there lies the success of rockets. During the war, the Germans had successfully realized the ME163. It was a fearsome interceptor and its fuel-to-total-weight ratio (W_f/W_g) was just 0.49. The SO-9000 Trident interceptor displayed at the Paris Salon of Aeronautics has been designed by the engineer Servanty in collaboration with the SEPR who developed the rocket engines. This truly supersonic airplane will make a fearsome interceptor, although its maximum sfc is probably close to 16.0 lb/lb, hr.

High temperature ramjets have a sfc of the order of 3.5 to 4.0 in the sonic range. However, in the supersonic range, $M = 2$ for instance, the sfc drops to 2.5, which corresponds to an efficiency of 21 per cent (close to that which is currently obtained with piston engine propeller combination). These relatively low sfc are due to the fact that adiabatic efficiencies are then close to unity because, fortunately, a shock wave is not always fatal.

The Future

We are now at the 600-mph stage. For civilian airplanes, the turboprop appears to have the advantage, but not necessarily for military airplanes, because even at subsonic speed the ramjet makes possible tremendous rates of climb.

Whether we like it or not, the interceptor must be capable of Mach 2 at high altitude. Without it, it looks to us difficult to intercept bombers doing Mach 0.95 at 50,000 ft.

In the supersonic range, aerodynamic performance is poor and it does not look as though great improvements will be made in the near future. The drag is enormous and L/D ratios ridiculously small. We have seen earlier the sea level equivalent thrust that had to be developed; namely, 143,000 lb for an airplane of 13,000 lb. In view of these requirements, it appears that the turbojet will be eliminated, leaving the competition to the ramjet and rocket engines.

After the first World War, when pursuit airplanes were doing 125 mph with difficulties, some thought that it would be difficult to fight at 200 mph. Presently we hear rumors that supersonic speeds are outside man's scope. Now, in my opinion, man is still far superior to any machine we can build.

But let us avoid the military field. He who would have said 50 years ago that today tourists would be comfortably carried at 500 mph over distances of 1200 miles in 50-ton airplanes would have been taken for a maniac or a joker.

Let's look again at the range equation

$$R = H_c \, J \eta_0 \, L/D \log_e \left(\frac{1}{1 - W_f/W_g} \right)$$

I have shown earlier that at $M = 2$ the over-all efficiency of the ramjet engine was the same as for propeller systems. This means that at that speed there is no propulsion system problem. In order to obtain the same range achieved by present airliners, it will be necessary to obtain at $M = 2$ values of the L/D of the order of 15 instead of 3 as we do currently. Is it really possible? I ask you. How about the next generations? Don't you think that they too will want to accomplish some progress in aerodynamics? And suppose they do not succeed: Is it not possible that tomorrow we may have available fuels more potent than petroleum fuels? I am sure that the last word has not been said; we already know of certain materials having a heating value which is practically infinite.

Man will not escape the devil nor the occult attraction of speed. This is why I have entire confidence in the future pre-eminence of ramjet engines.

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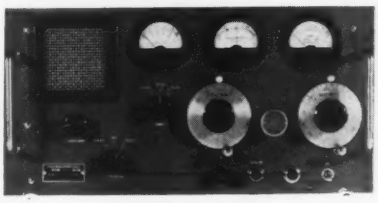

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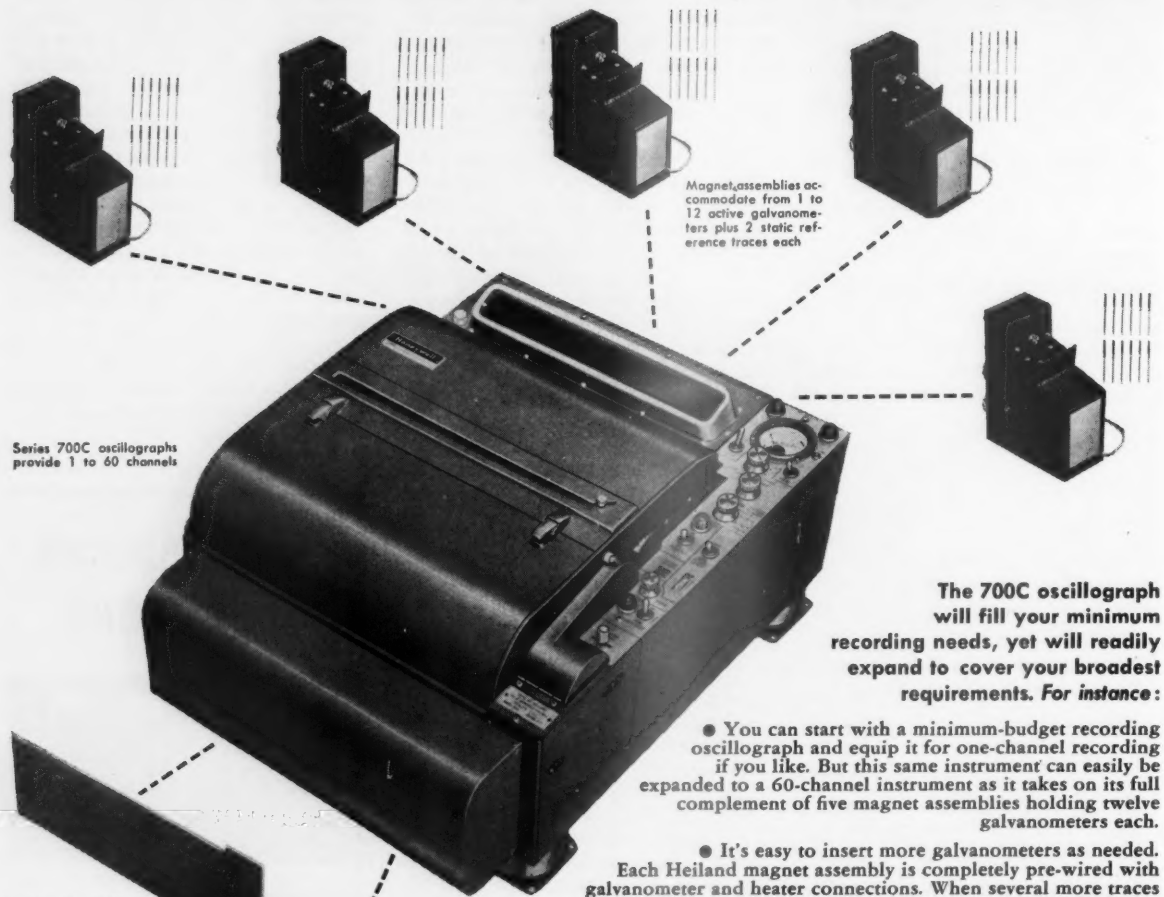
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Space Flight Notes

JOHN GUSTAVSON, Convair-Astronautics, Contributor

The Possibility of Interstellar Flight

Introduction

AFTER the recent opposition of the planet Mars, and now, when observations begin to yield results, it seems more doubtful than ever that we shall find life on other planets in our solar system.

The existence of algae and lichens, which we had hoped to find on Mars, may be a subject for discussion for many years yet to come.

The probability of finding life on Venus is very small; therefore, we must assume that the earth is the only life-sustaining planet in the solar system. The possibility of life existing in other systems has been contemplated by men over the ages; yet no reference can be made to any extensive treatment of the subject.

However, it does seem probable that one out of one hundred suns will host planetary systems (1).[1] There are 27 fixed stars within 13 light years from the sun (see Table 1); thus we have one chance in four to find other planets within this distance. When the right conditions prevail, life originates. This being so, we may find life in some manifestation less than a dozen light years away.

The nearest of the stars is Alpha Centauri Proxima which is 4.27 light years away. This very faint star can only be seen from the Southern Hemisphere. Alpha Canis Major, better known as Sirius, is 8.65 light years away. Sirius is a double star with a satellite of the White Dwarf class.

Transportation

Among the vehicles suitable for interplanetary operations, we find the chemical rocket with a specific impulse of 300–400 sec and an acceleration around 0.25 g; and the ion rocket with a specific impulse of 13,000 sec and an acceleration of 10^{-4} g.

The chemical rocket can immediately be dismissed as a means of interstellar transportation. A short powerful burst may give it hyperbolic velocity, but from then on, once it has attained this rate of motion, it will coast through space at only some ten miles per second. The journey to Alpha Centauri would take 70,000 years.

The ion rocket (2) can accelerate during the entire first half of the trip, decelerate during the second, and thereby shorten the time considerably. It is still, however, of the order of 1200 years. Within the next few decades development of nuclear pile techniques may well increase the acceleration to 10^{-3} g and thereby render a travel time of 400 years.

The chemical rocket converts 10^{-10} of the rest mass into propulsive energy, while modern atomic reactors convert 10^{-2} into energy. The ion rocket lies in between. A complete conversion of mass into energy after

$$E_0 = M_0c^2$$

is only known to take place in the electron-positron reaction, where the combined masses are converted into gamma radiation.

A photon rocket proposed by Saenger (3, 4) may, in the far future, employ light radiation propulsion. The exhaust velocity of the "combustion products," the photons, is equal to the highest attainable velocity, namely, the velocity of light.

Matter is fed into the reaction chamber and converted into radiation energy. The photons are reflected and directed out of the chamber by means of mirrors. The thrust from the photons, light quanta without rest mass, accelerates the vehicle.

For the crew of the photon rocket the trip to Alpha Centauri would seem to take 3.6 years if they started out with a mass ratio of 44; this means that the mass of the fuel, which is completely converted into energy as photons, is 43 times the mass of the empty rocket ship.

The crew, relying upon their chronometers and accelerometers, would experience an acceleration of 1 g over the entire flight time. Halfway to the star, the thrust direction would be shifted toward the target in order to decelerate the vehicle. Maximum velocity would be (as determined by the crew inside the ship) several times that of light. Seen from the earth, however, the space ship would accelerate less and less, only approaching the velocity of light, and the whole trip would seem to take 6 years.

At relativistic velocities, the interstellar or cosmic hydrogen gas will become a grave hazard to a vehicle;

Table 1 The nearest stars

	Distance		Relative
	parsec	light years	luminosity
Alpha Centauri A	1.32	4.3	1.15
Alpha Centauri B	1.32	4.3	0.3
Alpha Centauri C (Proxima)	1.31	4.3	0.00004
Barnard's Star	1.85	6.0	0.0004
Wolf 359	2.42	7.9	0.00002
Lalande 21185	2.61	8.5	0.005
α Canis Major A (Sirius)	2.65	8.6	27
α Canis Major B			0.0026
Ross 154	2.76	9.0	0.0004
Ross 248	3.13	10.2	0.00009
ϵ Eridani	3.30	10.8	0.28
61 Cygni A	3.31	10.8	0.055
61 Cygni B			0.03
α Canis Minor A (Procyon)	3.36	11.0	6.0
α Canis Minor B			0.0005
τ Ceti	3.40	11.0	0.34
Pos Med (Σ 2398) A	3.46	11.2	0.0031
Pos Med (Σ 2398) B			0.004
ϵ Indi	3.47	11.3	0.14
Lacaille 9352	3.69	12.0	0.012
Groombridge 34 A	3.70	12.0	0.006
Groombridge 34 B			0.0006
B.D. +5° 1668	3.73	12.1	0.0004
B.D. −12° 4523	3.74	12.2	0.001
Ross 614 A	3.80	12.3	0.0005
Ross 614 B			0.00008
Kapteyn's Star	3.86	12.6	0.0026
Lacaille 8760	3.92	12.7	0.03

[1] Numbers in parentheses indicate References at end of paper.

so grave, it is thought, that such velocities shall become prohibitive to flight.

A vehicle moving with near-light velocities through space will experience a bombardment of protons with energies equivalent to those of cosmic rays. The intensity will be much higher, and protective shielding apparatus would be extremely heavy.

From these considerations, we may conclude:

1 Interstellar flight will be possible by means of vehicles like ion and photon rockets.

2 Velocities must be kept low to avoid hazardous proton impact.

3 Travel time to and from the stars within a dozen light years from our solar system must exceed the lifetime of human beings.

Suspended Animation

The idea of prolonging the normal lifespan or achieving eternal youth seems to hold timeless fascination for us. The aging process in any living being has been studied by numerous investigators, and as many theories have been proposed. The trend shows a desertion of the hormonal theories while new emphasis is given to the study of the accumulation of dead cell material. But even by expecting a slow, steady increase in the lifetime of the average human being, we cannot visualize an interstellar mission carried out by the same original crew.

Two possibilities appear: allowing the crew to complete the voyage in a state of suspended animation, or providing for an expedition which would be genetically balanced to substantiate reproduction of the species in each generation. Both alternatives are very interesting but, unfortunately, out of reach of our present knowledge of physiology and sociology.

Suspended animation is a subject which has for some time intrigued the authors of science fiction stories. It is one theme which, of necessity, calls for more imagination than scientific facts. From a medical standpoint, suspended animation is characterized as a state of reduced metabolism. Thereby, the aging process is rendered almost negligible. Examples of lowered metabolism are found with the hibernating animals. The warm-blooded animals, however, like the bear keep their body temperature constant at normal level, and the metabolism is only slightly reduced.

Cold-blooded animals can survive a considerable lowering of the body temperature, and this is the basis of the few cases recorded on artificially created suspended animation. Frogs used for the ordinary pregnancy tests have been kept in refrigerators and have thereby had their metabolism lowered considerably. They hardly move and are extremely slow in all reactions. The reason for keeping them refrigerated is the abundancy of frogs during the summertime while relatively few can be found in the winter. After the "storage" in the refrigerator, they are slowly heated to normal temperature and then used for the test. An example of completely stopped metabolism, a true suspended animation, is the fast-freezing of live halibut fish, an experiment recently performed in Denmark. After thawing out, the halibuts showed all normal reactions.

These experiments, however, are only applicable for cold-blooded animals. Humans have been cooled to 82 F in order to slow down the blood circulation during heart surgery. The limit is set by the blood supply to the brain. The brain and all other organs have to be fast-frozen simultaneously to avoid cell rupture and later thawed simultaneously; even then we must count on the possibility of brain damage. A method for simultaneous thawing may be based upon high-frequency heating.

More promising is a chemically initiated state of suspended animation. The pharmacopoeia lists several drugs with somatic effect on the nervous system. Drugs instigating relaxation or complete immobility of muscles are listed. The best known of this group is probably curare, a series of isoquinoline-type alkaloids, such as curine, curarine, and tubo-curarine, which paralyze the ends of the motoric nerves. Curare was used as an arrow poison by the Campos Indians of South America. It is now used for anaesthetical purposes, as recovery from its effects is comparatively easy. The use of drugs for suspended animation is still purely theoretical.

The same can be said about reducing metabolism by electrical control of the brain. The present state of understanding of the function of the brain and nervous centers in the spine is still in its infancy.

Thus, we must conclude that suspended animation is not impossible, but certainly beyond our present knowledge of medicine.

Generations in Space

The number of children born to immigrant parents while on route to the United States is quite large; yet it takes imagination to envision the explorers who will give birth and die on their way to another star.

The ion rocket (acceleration = 10^{-3} g) will reach the nearest star in 400 years. One generation is 30 years; consequently, 13 generations will be born in space. The expedition has to be large enough to ensure genetical safety; that is, to avoid extensive intermarriage. On the other hand, an expedition should be limited in size to keep the over-all weight, volume, and power requirements low. A complete regenerative cycle to avoid any losses of matter from the interstellar vehicle must be employed.

Besides the technical problems, we find also the sociological complications. The first generation starting out from the earth will have the knowledge of never returning, while their children will be brought up within the confines of the space ship, hearing only of the earth as the home planet of their parents. There will be, no doubt, people who will volunteer for an eventual interstellar vehicle. Shepherd (5) points out that "...the philosophy of the explorer may be that of the soldier or airman setting out on a suicide raid, doing so in the knowledge that for him there can be no personal gain, only the dying knowledge that some will survive to benefit from his action." I feel this is somewhat of an exaggeration. The explorers will feel, rather, like immigrants leaving the Old Country with the knowledge of never seeing it again but, at the same time, looking forward to their new life. The explorer will look forward to the development of life in the ship, a life which may take a different course from that of the earth. Communication difficulties will make exchange of larger amounts of information impossible, thereby prohibiting exchange of philosophy. The inhabitants in the confined space of the ship will develop independent of earthly influence.

This cursory introduction shows that interstellar flight is well beyond our present knowledge and technology. The theoretical possibility prevails, and men will eventually travel to other solar systems.

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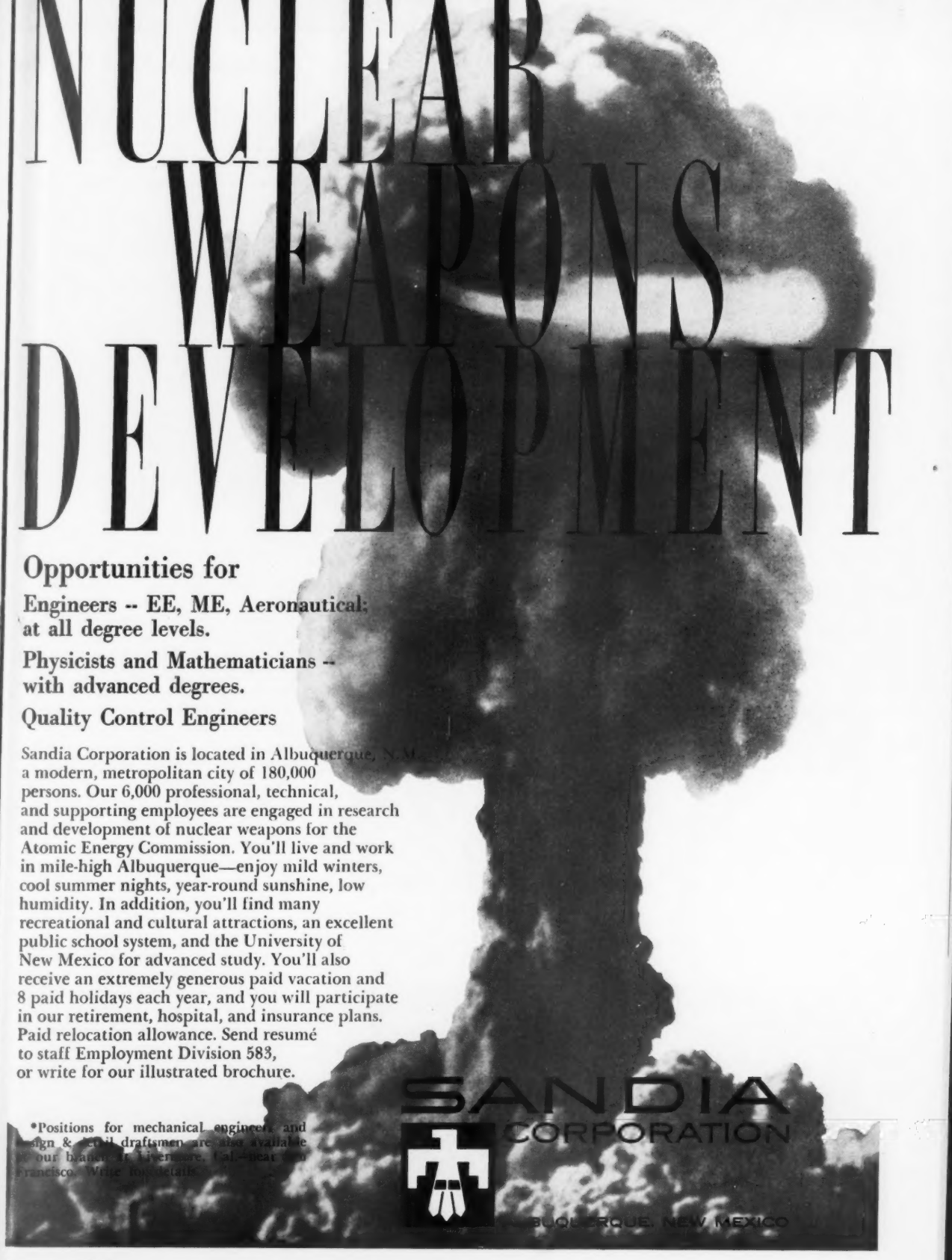
NUCLEAR WEAPONS DEVELOPMENT



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Past and Present Mix at Directors Reception

In addition to elevating Commander Robert C. Truax from last year's vice-presidency to this year's presidency, members of the AMERICAN ROCKET SOCIETY also elected George P. Sutton as vice-president and voted Krafft A. Ehricke, Harold W. Ritchey, and John P. Stapp to three-year terms on the Board of Directors, replacing George Sutton and J. B. Cowen. (Dr. Ritchey, completing a two-year term, was chosen to succeed himself for a full three-year term. For a complete list of the new Board, see masthead.)

During the 11th Annual Meeting, the Directors Reception on November 29 afforded many of the new and current officials of the Society the opportunity to begin or renew acquaintances with those who have served in the past.

* Dr. and Mrs. Porter (center) and Dr. and Mrs. Davis (right).

Incoming vice-president Sutton (center) is briefed by past President Davis (right) and IAS President Edward Sharp.

Director and Mrs. Wernher von Braun huddle with Dr. von Braun's chief, Major General John B. Medaris.

Two old hands, Kurt Stehling and Noah Davis, give President Truax some humorous, if not significant, advice.

Truax introduces Maj. Gen. Schriever to honorary members Harry F. Guggenheim and Esther C. Goddard.

Annual ARS Meetings

THERE was still a month to go in 1956 when the SOCIETY's four-day Annual Meeting closed November 29. But all indications pointed backward to a highly successful year under President Noah S. Davis.

More important, from trends now apparent, Commander Robert C. Truax, newly installed president of the SOCIETY, can look forward to an even more successful year in 1957. Some significant signs, as disclosed at the annual business meeting on Nov. 27:

- Membership, individual and corporate, hit a record high last year and promises to maintain the pace in 1957.
- Income and net worth of the SOCIETY are also at an all-time high and show no signs of abating.
- JET PROPULSION, the official Journal of ARS, is expanding at a tremendous clip, published 400 more pages in 1956 than in 1955—over a 50 per cent increase.

And . . . This was only part of the picture. At the meeting itself, held in New York City from Nov. 26–29, over 1200 people attended, listened to 48 different papers covering almost every aspect of jet propulsion (see Sessions & Subjects p. 75).

Held in conjunction with the annual meeting of The American Society of Mechanical Engineers, the 11th Annual Meeting of the AMERICAN ROCKET SOCIETY also offered registrants the opportunity to attend pertinent ASME sessions as well as the special Rockets and Missiles Exhibit, sponsored by ARS as part of the 22nd National Exposition of Power and Mechanical Engineering. Then, on Nov. 28, there was the New York Section Film Night which proved to be a favorite attraction, drawing over 700 people to a viewing of the latest documentaries on missiles and rockets.

On the business side, a final count of ballots showed that members—in addition to electing new officers (see box: Directors Reception)—had approved three amendments to the SOCIETY's bylaws. Effective the first of this month, the amendments, in brief, (1) raise the dues of Associate Members to $15/year, and (2 & 3) provide in the bylaws for the establishment of student chapters.

Grass Roots Report

Sandwiched between four full sessions on Nov. 27 and lying in the shadow of the Honors Night Dinner on Nov. 29 (see right), the ARS Section Luncheon

...ps A Successful Year

HAIL AND FAREWELL: Noah S. Davis passes president's gavel to Robert C. Truax.

did not get the full attention it merited from members.

At the luncheon, presided over by Dr. Davis and attended by 140 people, officers of different sections reported on activities of their local group during 1956 and plans for 1957. This roundup served to show not only a substantial rise in the number of Sections; but, even more important, it brought to light a significant increase in the scope and significance of Section activities. Items:

• At the time of the 1956 Annual Meeting there were 31 Sections in existence, three being formed. In November 1955, there were 23 Sections.

• Almost every Section reported more meetings in 1956 than in 1955, planned to have an even more active program in 1957.

• At least seven Sections established and were following through on active educational programs designed to interest students in scientific careers. These programs, of course, particularly emphasize careers in the jet propulsion field and try to interest students in ARS activities.

• Most sections are also making strong membership drives. Some reported that they had more than doubled their enrollment in 1956, hoped to equal this feat in 1957.

Definitely optimistic, reports from the ARS Section Luncheon added substance both to last year's recorded success and to this year's anticipated growth of the AMERICAN ROCKET SOCIETY.

Eleventh Annual Honors Night Dinner

For Past Achievements, A Night of Recognition

GAINING momentum for four days, the 11th Annual Meeting approached the Honors Night Dinner on November 29 at a fast clip. And, somewhat as expected, the Dinner proved to be the high spot of the annual event, a fitting climax to a full four days.

In an unusual display of strength, approximately 700 people came to pay tribute to leaders in the field of jet propulsion. A Janus-faced affair, the 1956 presentation of awards and fellowships reached all the way back to 1923 to honor Hermann Oberth for his book, "Die Rakete zu den Planetenraumen," and then reached into the future in granting recognition to the potential of 17-year-old basement-rocketeer James Blackmon. (For a complete rundown on award recipients, see 1956 Honor Roll.)

And while it was bestowing individual recognition, the AMERICAN ROCKET SOCIETY was itself the recipient of a collective tribute from Rear Admiral James S. Russell, chief of the Navy's Bureau of Aeronautics and featured speaker at the Honors Night affair.

"I know of no single independent group of people," said Admiral Russell, "whose accomplishments can match those of the American Rocket Society, which has done so much and helped so materially in creating a major science and industry."

Headlines Too: Admiral Russell went on to discuss rocket power as used in the Navy—past, present, and future. In doing so, he produced the following news-making disclosures:

• The Navy now has 10 ships, including submarines, equipped with the Regulus; 2 cruisers, Boston and Canberra, armed with the Terrier; and the first Talos ship will be ready in 1958. In five years, said Admiral Russell, the Navy will have at least 8 Talos cruisers, 22 Terrier ships, and 17 Tartar ships.

• On the subject of military aviation: In its next generation of fighters, the Navy will use rocket motors to supplement turbojet propulsion. Future engine designs may even combine rocket and turbojet functions in a single power plant package.

• Admiral Russell indicated that the Navy was already experimentally flying rocket-turbojet combinations.

In closing, Admiral Russell stressed the importance of the contributions that the military could make as a member of the rocketry team.

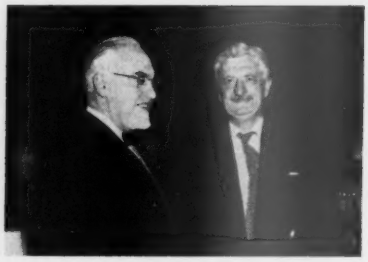
ADMIRAL RUSSELL accents role of Navy and ARS in jet propulsion progress.

1956 Honor Roll

Society Honors Seven Awards Winners and . . .

Chandler C. Ross receives Robert H. Goddard Memorial Award from Mrs. Goddard.

Bruce H. Sage (left) accepts C. N. Hickman Award from Dr. Hickman.

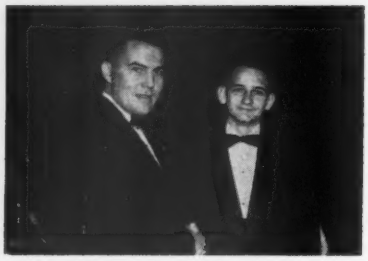


Dr. Pendray (left) presents G. Edward Pendray Award to Hermann Oberth.

Joseph Kaplan gets ARS Astronautics Award from Andrew G. Haley (left).

Mrs. Wyld presents James H. Wyld Memorial Award to Louis G. Dunn.

James Blackmon receives ARS-Chrysler Award from Maj. Gen. John B. Medaris.



Jack B. Cowen gives ARS Student Award to Donald L. Crabtree.

. . . Selects Seven New Fellow Members

Charles W. Chillson

William C. House

Maj. Gen. Bernard A. Schriever

Russell K. Sherburne

EDITOR'S NOTE: Unable to be present, Major General Donald N. Yates, Simon Ramo, and William H. Pickering delegated others to accept Fellow Member Awards for them.

Sessions & Subjects

HEART of any scientific society meeting—annual or otherwise—is, of course, the individual paper. At the November meeting, there were 48 papers presented. Many of those from the two sessions on High Altitude Sounding Rockets have already been reported in some detail (see JET PROPULSION, Dec. 1956, p. 1109). Others will be covered, wholly or in part, in future issues of the JOURNAL.

But here, for the sake of easy reference, is a rundown on all papers presented at the 11th Annual Meeting, grouped according to the sessions in which they appeared:

Solid Propellants

✦ An Experimental and Theoretical Study of the Burning Mechanism of a Heterogeneous Propellant Based on Ammonium Perchlorate, by Martin Summerfield and George Sutherland, Princeton University. (360-56)

Research was undertaken to clarify the burning mechanism of composite solid propellants. From this emerged a theory of steady state burning which led to the burning rate law $1/r = a/p + b/p^{1/3}$. Conclusion: The burning of a certain class of heterogeneous propellants is controlled by the rate of reaction in a so-called granular diffusion flame zone of the type observed in these experiments.

✦ Petroleum Based Solid Rocket Propellants, by J. A. McBride and R. W. Scharf, Phillips Petroleum Co. (338-56)

Authors cover development, properties, and applications of JATO, 16-NS-1,000, M15, the first solid propellant charge (and motor components) to comply completely with stringent USAF specifications for a rocket engine.

✦ Some Properties of a Simplified Model of Solid-Propellant Burning, by Leon Green, Jr., Aerojet-General Corp. (339-56)

Green examines behavior of a model of propellant burning in which a gross simplification of the burning mechanism is attempted. His analysis indicates that the simplified thermal model exhibits both steady and nonsteady behavior consistent with the known burning characteristics of solid propellants.

✦ Some Preliminary Photoelastic Design Data for Stresses in Rocket Grains, by D. D. Ordahl of Naval Ordnance Test Station (China Lake, Calif.) and M. L. Williams of California Institute of Technology. (340-56)

From an investigation of quantitative relationships between the various parameters of grain geometry (for internal-burning star-perforated grains) and the elastic stress distribution in the propellant, authors prepared engineering design charts that can be used to determine maximum stress developed in any prescribed configuration within the family investigated providing the properties of the propellant are in the elastic range.

Space Law and Sociology

✦ The Rate Processes of Life and the Special Relativity Theory, by William Brewster, Jr., Harriet Brewster, Majic Potsaid, and James Isaacs, Harvard University. (376-56)

The idea that space crews will return to earth younger than they would have been had they stayed at home, the Harvard group declares, is a fallacy exploited by science-fiction writers but exploded by consideration of Einstein's special theory of relativity.

✦ Projecting the Law of the Sea into the Law of Space, by Rear Admiral Chester Ward, USN. (377-56)

Ward discusses some aspects of the law of the sea that may serve as a possible source of useful principles that may ultimately find application in the law of space. But, he warns, "you can't legislate without facts."

✦ The Present Day Developments in Space Law and the Beginnings of Metalaw, by Andrew G. Haley of Haley, Doty & Wollenberg. (378-56)

This is a report on latest developments in space law of current import (e.g., the legal status of the upcoming satellite launching) and a discussion of metalaw—the law governing the rights of intelligent beings of different natures and existing in an indefinite number of different frameworks of natural law.

✦ Some Social Implications of Space Travel, by Colonel William Davis, Air Research & Development Command. (379-56)

Paper centers on the implications of space flight for the social anthropology of small groups, brings to light significant data from World War II research and underscores need for further research.

High Temperature Materials

✦ Molybdenum for High Strength at High Temperatures, by Robert Freeman and J. Z. Briggs, Climax Molybdenum. (341-56)

Properties of molybdenum and its alloys along with their applications (actual and potential) in the jet propulsion field are covered in this paper.

✦ Temperature Indicating Paints, by J. C. Becker, Aerojet-General Corp. (342-56—There will be no preprints of this presentation which consisted in large part of colored movies and slides.)

In a search for a good temperature-indicating paint for use in determining hot spots of missiles and rockets, Becker uncovered a paratertiary-phenol-formaldehyde varnish (described in BuAir Bulletin 66) with an accuracy of ±50 F over a range 300 F–900 F.

✦ Structural Materials for Missile Applications at Very High Temperatures, by J. R. Kattus, Southern Research Institute. (364-56)

For use in the design of missile components exposed to aerodynamic heating, Kattus offers data on some short-time mechanical properties of copper, iron, molybdenum, tantalum, and graphite at high temperatures.

✦ Graphite as a Material for High Temperature Gas Flow Systems, by Leon Green, Jr., Aerojet-General Corp. (382-56)

Author reviews mechanical properties of graphite, points out advantages and limitations of the substance as a fabrication material for high temperature gas flow systems.

Combustion

✦ Effect of Turbulence on Radiation Intensity from Propane-Air Flames, by Richard John and Martin Summerfield, Princeton University. (343-56)

To determine extent of possible interaction between mixing process and chemical reaction, authors measured the effect of turbulence on flame radiation intensity. Their results indicate that turbulent mixing breaks up what would be a continuous laminar flame sheet and that this breakup modifies the chemical reaction and transport processes in the flame zone.

✦ Analysis of Combustor Performance Based on Simplified Chemical Kinetics, by Ernest Mayer, Arde Associates. (344-56)

In this paper, Mayer presents an analysis of the performance of a piloted can-type burner as a representative model of combustors with mass addition and heat evolution governed by a reaction rate law.

✦ An Investigation of the Behavior and Reaction Mechanics of Nitric Acid-Hydrocarbon Flames, by M. H. Boyer and P. E. Friebertshauser North American Aviation, Inc. (371-56)

The research presented in this paper concerns a study of Bunsen flames supported by mixtures of nitric acid and hydrocarbon vapors.

✦ The Use of Mollier-Type Thermodynamic Charts for Calculating Performance and Design Parameters, by C. H. Trent, S. R. Goldwasser, and J. M. Carter, Aerojet-General Corp. (372-56)

Authors have prepared a set of Mollier charts plotting specific enthalpy vs. pressure for the gaseous products resulting from combustion of liquid oxygen and JP-4 for pressures ranging from 0.25 psia to 1000 psia and for mixture ratios 1.7, 1.9, 2.1, 2.3, 2.5, and 2.7. And there are auxiliary graphs, as well, representing the variation in combustion gas average molecular weight with both pressure and temperature.

✦ Industrial Applications of the Rocket Principle, by F. R. Job, Linde Air Products Co. (350-56)

Paper covers industrial development of the rocket principle which has taken place primarily in the mining and quarrying fields.

High Altitude Sounding Rockets—I

✦ The High Altitude Sounding Rocket, by Homer Newell, Jr., Naval Research Laboratory. (345-56)

✦ Aerobee-Hi Development Program, by John Townsend, Jr., of Naval Research Laboratory and Robert Slavin of Air Force Cambridge Research Center. (346-50)

✦ The Nike-Cajun Sounding Rocket, by Leslie Jones and Nelson Spencer of University of Michigan, William Stroud of Evans Signal Laboratory, and Warren Berning of Aberdeen Ballistic Research Laboratory. (361-56)

✦ Future Sounding Rockets, by W. C. House, C. H. Dodge, R. D. Waldo, A. Schaff, Jr., J. L. Fuller, and O. J. Demuth, Aerojet-General Corp. (347-56)

(For details of these papers see JET PROPULSION, Dec. 1956, p. 1109).

Liquid Propellant Rockets

✦ Importance and Application of Thrust Control for Liquid Propellant Rockets, by Rudolf Reichel, Bell Aircraft Corp. (348-56)

The purpose of this paper, says Reichel, is to present an introduction to thrust control engineering problems as involved in various rocket power plant applications and not to specialize on analytical treatments (i.e., flight—as well as control dynamics) which represent a very special field.

✦ Dynamic Characteristics of a Liquid Filled Tube, by W. A. Sibley and W. G. Oakes, Boeing Airplane Co. (349-56)

Authors report on their investigation to define certain elastic wave phenomena associated with oscillating pressures in a liquid filled line.

✦ An Approximate Theory for Discharge Coefficients of Flow Nozzles, by F. S. Simmons, North American Aviation, Inc. (351-56)

Paper offers the derivation of an analytic relation among discharge coefficient, Reynolds number, and nozzle geometry by utilization of elementary boundary layer theory.

✦ A Philosophy for Improved Rocket Nozzle Design, by Robert Dillaway, North American Aviation, Inc. (362-56)

Dillaway presents a three-dimensional analysis of supersonic contoured rocket

nozzles designed to arrive at the design of a shortened rocket nozzle of high performance.

✦ First Stage Propulsion—Project Vanguard, by George Macpherson, General Electric Co. (385-56)

Presentation describes GE's role in building the first-stage power plant for the three-stage earth satellite vehicle.

High Altitude Sounding Rockets—II

✦ ASP (Atmosphere Sounding Projectile), by Charles Zimney, Cooper Development Corp. (352-56)

Zimney details many of the design, performance, and application aspects of this relatively new single-stage solid propellant sounding rocket.

✦ The Navy Rockaire Program, by Lieut. Commander Malcolm Ross, Office of Naval Research. (380-56)

Ross' paper covers the development of the Navy's interest in sounding rockets launched from balloons and aircraft.

✦ The Air Force Rockaire Program, by Robert Slavin, Air Force Cambridge Research Center. (381-56)

Slavin reports on the development and present status of the USAF program dealing with balloon- and aircraft-launched high altitude sounding rockets.

✦ The French Missile—"Veronique," by E. Vassy, University of Paris. (353-56)

(See JET PROPULSION, Dec. 1956, p. 1110.)

✦ Japanese Sounding Rockets, by Hideo Itokawa, University of Tokyo. (363-56)

Author covers the development and performance of various sounding rockets developed by Japan for use in upper air research.

✦ British High Altitude Rocket Research Programme, by E. B. Dorling, Royal Aircraft Establishment. (373-56)

(See JET PROPULSION, Dec. 1956, p. 1110.)

Atomization and Sprays

✦ Instrumentation to Measure Composition and Temperature of High-Velocity, Two-Phase, Two-Component Flows, by K. R. Wadleigh and R. A. Oman, Massachusetts Institute of Technology. (354-56)

This paper describes the current state of development of probes designed for three distinct measuring functions: (1) local water vapor-air mass flow ratio, (2) local total water-air mass flow rates, and (3) local gas phase stagnation temperature.

✦ Theoretical and Experimental Study of the Deformation and Atomization of a Liquid Drop in a High Velocity Gas Stream, by Naotsugu Masugi, Massachusetts Institute of Technology. (355-56)

The second portion of the over-all process of gas atomization—the breakup of large drops into smaller ones—is analyzed in this paper.

✦ The Atomization of Liquid by Means of Flat Impingement, by Yasusi Tanasawa, Tohoku University. (356-56)

This is an experimental study of the atomizing characteristics of flat (180 deg) impingement nozzles. From the results, the author proposes a new type of impingement nozzle with flat spray and controlled flow rate.

Rocket Production Techniques

✦ Important Factors of Rocket Engine Development with a Foresight on Mass Production, by Guenther Haase, Bell Aircraft Corp. (365-56)

In his presentation, the author shows how the purpose of a missile as well as mass production requirements influence basic missile design.

✦ Redstone Missile Assembly Problems, by Charles Williams, Chrysler Corp. (366-56)

Williams describes some of the fabrication and assembly problems that Chrysler, as an automotive manufacturer, faced in entering the engineering development and manufacture of a large ballistic missile.

✦ Design and Production of the Falcon Solid Propellant Rocket Motor Case, by Carl Johnson, Scaife Co. (368-56)

Paper deals with the development phases of the Falcon rocket motor case over the past six years.

Space Flight Symposium

✦ Heat Transfer to Satellite Vehicles Reentering the Atmosphere, by N. H. Kemp and F. R. Riddell, Avco Research Laboratory. (358-56)

Authors apply a new theory of heat transfer at high velocities and high altitudes to the problem of a satellite vehicle entering the atmosphere.

✦ Some General Considerations of the Utility and Operation of a Long Range Manned Rocket Research Vehicle, by G. Harry Stine, White Sands Proving Grounds. (357-56)

Stine concerns himself with feasibility of a long-range manned rocket research vehicle.

✦ Lifetimes of Artificial Satellites of the Earth, by Irvin Henry, Aerojet-General Corp. (359-56).

(Paper published in this issue of JET PROPULSION, p. 21.)

✦ Skin Temperatures of a Satellite, by Craig Schmidt and Arnold Hanawalt, Bell Aircraft Corp. (383-56)

This presentation is concerned primarily with determination of surface temperatures on satellite vehicles.

✦ A New Type of Nuclear Power for Space Flight, by Robert L. Carroll, Naval Air Test Center. (375-56)

Carroll conceives a new type of nuclear reaction that he believes could find application in powering of space ships. As described by the author, it is a "cold" process; i.e., it would absorb heat, but would result in the liberation of high velocity nuclear mass particles.

Reliability

✦ Reliability Concepts in Rocket Power Controls Design, by H. L. Coplen, Aerojet-General Corp. (369-56)

Main point of this discussion is that maximum rocket engine reliability can be achieved only by a control system having functional simplicity as a result of real compatibility of the major engine components and the use of all inherent self-controlling elements in the system.

✦ Reliability Measurement by Machine Methods, by Richard Landers, General Electric Co. (367-56)

Landers evolves a program for measuring reliability of a weapons system which employs machine methods for punching, sorting, collating, analysis, computation, and tabulation.

✦ Sequential Analysis as a Practical Method of Satisfying Reliability Requirements, by William Brewington and Bernard Tiger, Reaction Motors, Inc. (384-56)

Primary goal of this paper is to determine and define a statistical method which can be used to demonstrate the reliability of a given system.

✦ Designing Equipment for Reliability, by R. B. Wilson, Convair. (374-56)

Culled from experience, a list of design check-points is offered by Wilson along with a program aimed at bringing to light critical weaknesses in aircraft and missile equipment.

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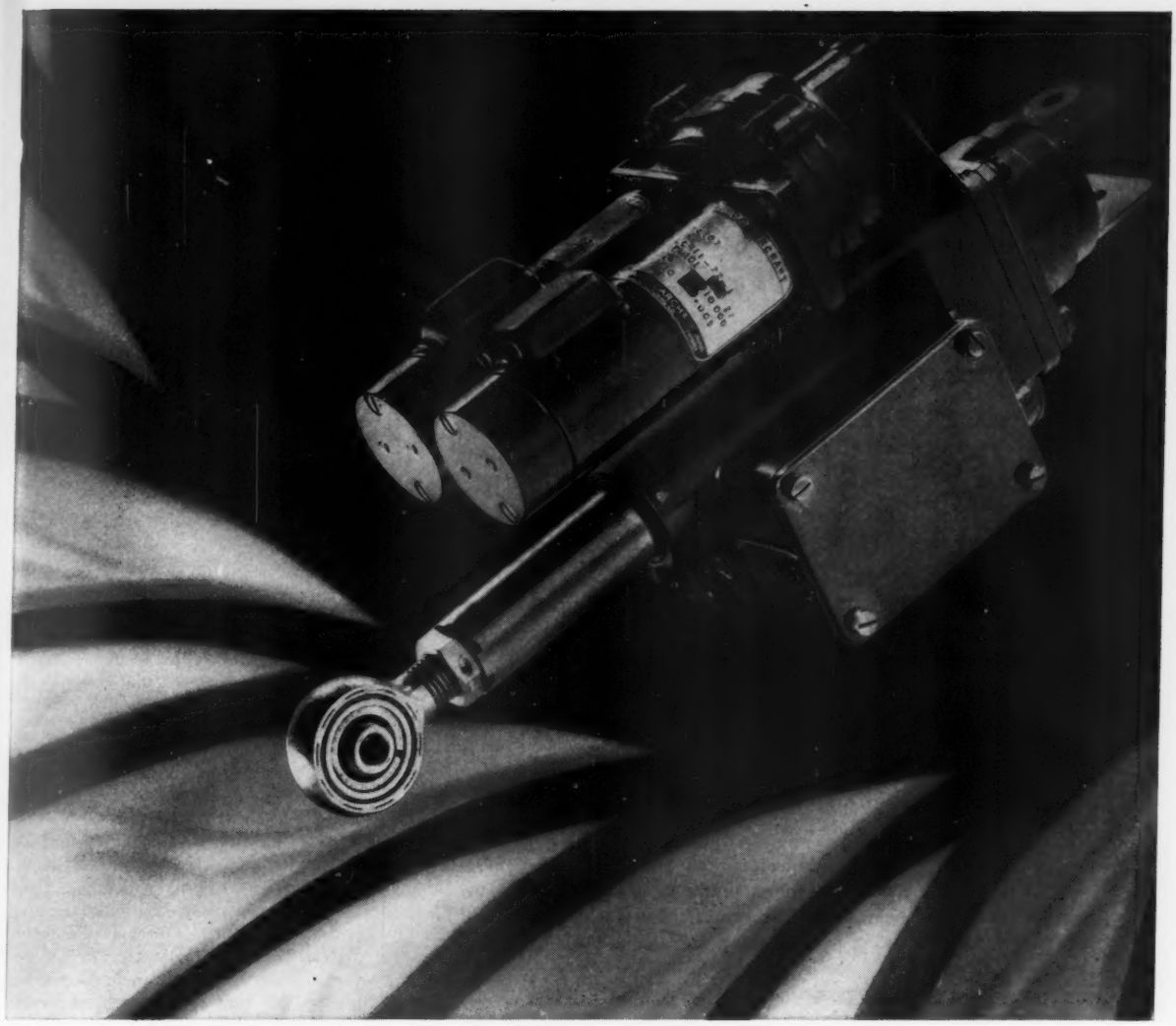

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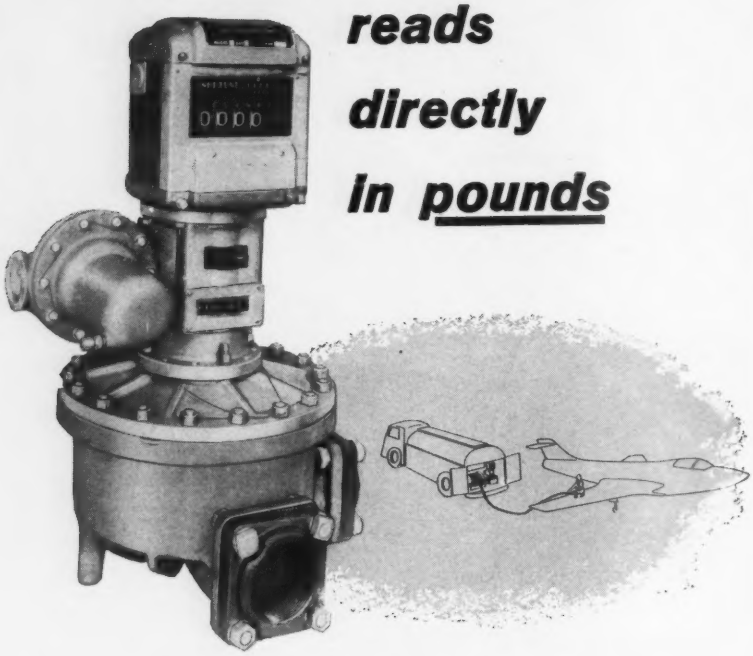

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Sections

Columbus: Abbott Putnam, Battelle Memorial Institute, covered screech and other oscillatory phenomena in a talk at the October meeting.

Detroit: A joint gathering of the Detroit Section and the Engineering Society of Detroit on September 19 listened to Dr. Hubertus Strughold, chief of the Department of Space Medicine at Randolph Air Force Base, discuss the anticipated physical rigors of space flight.

Charles W. Williams, former Section vice-president, has moved up to the presidency in an election that also raised former treasurer Lovell Lawrence, Jr., to the vice-presidency and brought in C. Tait as treasurer and Louis B. Forman as secretary.

Indiana: Approximately 60 people attended the November 1 meeting at Purdue University, listened to Donald Bartz of California Institute of Technology present "A Survey of Rocket Motor Heat Transfer Problems."

On November 13, Donald L. Crabtree (see p. 74), in the first of the section's series of seven space flight seminars, gave a short history of the development of rockets.

Maryland: Elected as officers for 1957 are: Andrew W. McCourt, president; Samuel Fradin, vice-president; Walter J. Crotty, Jr., secretary; and Edward T. Munnell, treasurer.

Approximately 70 people attended the November 13 dinner meeting. Host for the occasion was Fairchild Engine and Airplane Corp. Guest speaker Sears Williams of the Martin Co., in his lecture on the satellite launching vehicle, pointed out that error tolerances on the launching vehicle were so tight that they precluded any hope of a circular orbit.

Sacramento: On October 9, David Sprenger of Aerojet spoke to Section members on design and development aspects of large solid propellant rockets.

Section President Frank Coss has resigned because of a job change that necessitated his leaving the area. Daniel Tenenbaum, vice-president, has taken over Mr. Coss's functions.

New York: November meeting—held in conjunction with ARS Annual Meeting—featured a "film night" on the 28th.

ARS Meetings Calendar

April 3–5: ARS Spring Meeting, Sheraton Park Hotel, Washington, D. C.
June 10–13: ARS Semi-Annual Meeting, Hotel St. Francis, San Francisco.
Aug. 25–28: ARS-Northwestern Technological Institute Gas Dynamics Symposium, Northwestern University, Evanston, Ill.
Dec. 2–6: ARS Annual Meeting, New York.

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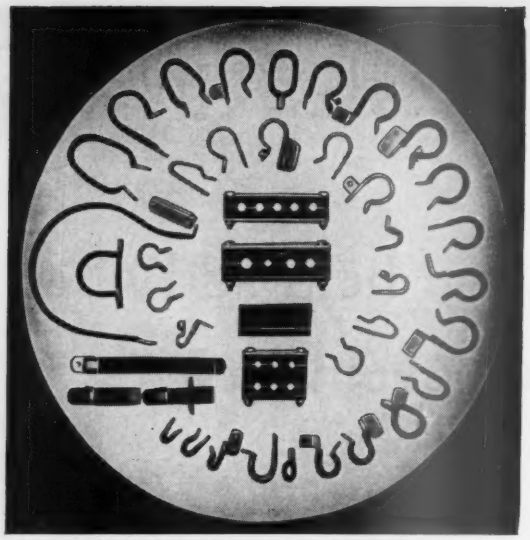

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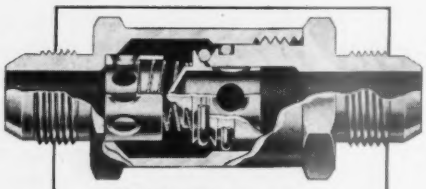

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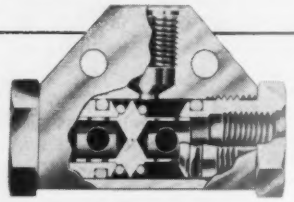

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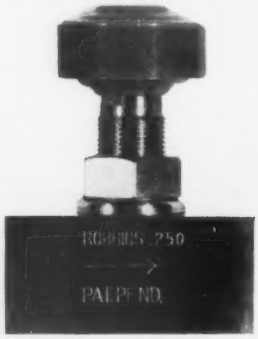

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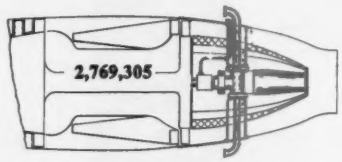
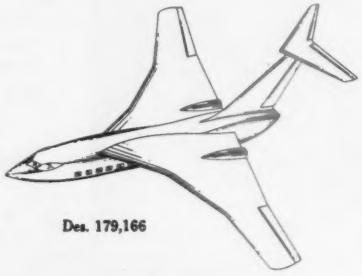


Power plants comprising main and auxiliary engines (2,769,305). George Chernowitz, Caldwell, N. J.

Rocket engine mounted within the forward section of a jet engine. Rocket exhaust gases discharge into (and exercise an ejector action on) the jet engine exhaust gases.



Des. 179,166

Airplane design (179,166). Walter Tydon, Hagerstown, Md., assignor to Fairchild Engine and Airplane Corp.

Jet deflector having curved guide wall for aircraft (2,770,095). Peter F. Ashwood and Philip J. Fletcher, Farnborough, England, assignors to Power Jets (Research and Development).

A guiding boundary wall extends outside the jet nozzle, curving round into a direction transverse to the direction of its walls. The jet can be controlled to separate from the wall or follow its curvature into the transverse path.

Continuous combustion chamber with air flow and turbulence controlled by flame front position (2,770,096). Homer M. Fox, Bartlesville, Okla., assignor to Phillips Petroleum Co.

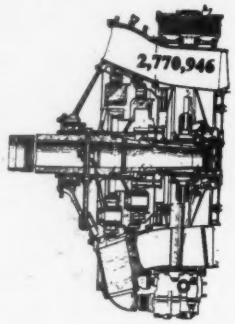
Fuel-air ratio in the primary combustion zone is varied in response to the position of the flame front in the burner tube.

Variable area reaction nozzle movable from convergent-divergent form (2,770,944). Donald J. Jordan, Glastonbury, Conn., assignor to United Aircraft Corp.

Compressed air actuated piston moves flaps inwardly for optimum thrust generation during nonafterburning operation. Flaps are moved outwardly during afterburner operation when the exhaust gases are supersonic.

Jump-type flow divider for gas turbine power plant fuel system (2,770,945). Hugh S. Crim, Glastonbury, Conn., assignor to United Aircraft Corp.

Fuel system with two manifolds, and a fuel divider with pressure responsive means for regulating the admission of fuel to the secondary manifold.



Brake for turbine rotor (2,770,946). Bennett S. Savin, Jr., Glastonbury, Conn., assignor to United Aircraft Corp.

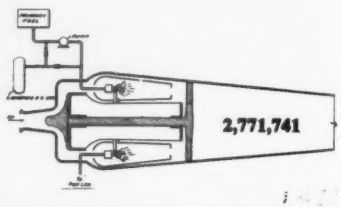
Power plant with high and low pressure rotors forming independently rotatable spools. A gear connection from one spool to a brake stops rotation of the spool, and a starting means is provided for rotating the other spool.

Jet power system (2,771,257). Donald B. Doolittle, Wilmington, Del., assignor to All American Engineering Co.

Jet engine and a rotor with a hollow hub to receive the exhaust through a fixed hollow dome with at least one opening. Rotor supported tailpipes have nozzles which pick up exhaust from the hub and impart tangential thrust to the rotor.

Morpholines as hypergolic fuels (2,771,738). C. R. Scott and A. L. Ayers, Bartlesville, Okla., assignors to Phillips Petroleum Co.

Thrust produced in a reaction motor by introducing into the combustion chamber an oxidant and a fuel comprising a morpholine consisting of a hydrogen atom and a hydrocarbon radical from the group consisting of alkyl, alkenyl, aryl, cycloalkyl, cycloalkenyl, aralkyl, and alkaryl radicals.



Low-temperature starting of gas turbine engines (2,771,741). Daniel P. Barnard, 4th, Chesterton, Ind., assignor to Standard Oil Co.

Method comprises storing in a closed pressure zone a liquefied normally gaseous

EDITOR'S NOTE: The patents listed above were selected from recent issues of the Official Gazette of the U. S. Patent Office. Printed copies of patents may be obtained at a cost of 25 cents each, from the Commissioner of Patents, Washington 25, D. C.

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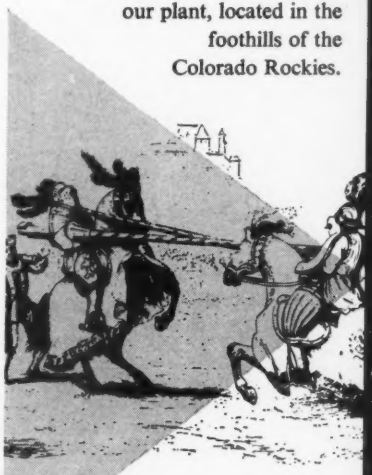

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MARTIN
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hydrocarbon having a vapor pressure of about 15 psi to about 100 psi at a temperature of −100 F. Hydrocarbon is introduced into the jet fuel during the starting period.

Gas turbine engine with reheat combustion equipment (2,771,743). A. C. Lovesey, Littleover, England, assignor to Rolls-Royce, Ltd.

Means for injecting air in jet form penetrating the high velocity gas stream at right angles to the direction of the gas flow.

Fuel injection means of combustion equipment for gas turbine engines (2,771,744). F. M. Johnson and D. R. Carlisle, Risley, England, assignors to Rolls-Royce, Ltd.

Air flowing into the combustion space through a passage is diverted outwardly adjacent to the outlet passage where fuel is supplied to a receiving surface.

Launcher (2,771,811). Charles C. Lauritsen, Pasadena, Calif., assignor to the U. S. Navy.

Pair of I-section rails, the upper and lower flanges defining tubular portions of triangular cross section and slots for accommodation of a launching lug of each rocket to be launched.

Rocket launcher (2,771,818). Charles C. Lauritsen, Pasadena, Calif., assignor to the U. S. Navy.

Launcher for rockets for movement from a stowed position resting on a deck to a raised position for loading and firing. All parts are out of contact with the deck. In the loading position, the launching rail is spaced from the deck so as to permit loading from the rear even while the rail is in a position of extreme elevation.

Air inlet for supersonic airplane or missile (2,772,620). Antonio Ferri, Queens County, N. Y.

Flow passage in line with the air stream having means to accept and decelerate the air with low external compression and high pressure recovery. At least one vane extending across the passage at the inlet end has a streamlined bulge intermediate to its upstream and downstream edges, forming a convergent-divergent passage.

Rocket propulsion method (2,771,739). Frank J. Malina and John W. Parsons, Pasadena, Calif., assignors to Aerojet-General Corp.

Red and white fuming nitric acids, and liquid fuel consisting of lithium, beryllium, boron, aluminum, magnesium, phosphorous potassium, sodium and mixtures thereof, dispersed in liquid hydrocarbon.

Afterburning means for turbojet engines (2,771,740). Clarence L. Johnson, Encino, Calif., assignor to Lockheed Aircraft Corp.

Fuel injecting means movable from retracted position to a position projecting into the tailpipe.

High altitude observation means (2,763,447). Mario J. Carrau, Los Angeles, Calif.

A helicopter within the frame of a rocket launched into space. Jet engines on the rotor tips maintain the helicopter in flight when it is released from the rocket through a detachable head.

Air-fuel ratio control system for ram-jet engines (2,765,619). Joel D. Peterson, Riverdale, N. J., assignor to Bendix Aviation Corp.

A predetermined airflow is effected by changing the datum of the regulating means to vary the selected temperature and combustion chamber inlet flow pressure.

Gas turbine apparatus (2,766,580). Wayne W. Buerer and Robert A. Neal, Media, Pa., assignors to Westinghouse Electric Corp.

Actuating means responsive to a predetermined engine speed connected to a normally open variable area exhaust nozzle adapted to effect reduction in the flow area.

Detection and prevention of combustion instability in jet engines (2,766,583). Robert M. Schirmer, Bartlesville, Okla., assignor to Phillips Petroleum Co.

Thermocouple, in the combustion chamber, having a characteristic response time of less than $^1/_{10}$ sec. Measurement is made of the alternating-current component of the thermocouple output while filtering the direct-current component.

Retractable afterburner (2,768,498). Harvey C. Karcher, Indianapolis, Ind., assignor to General Motors Corp.

An annular plate, downstream of a tubular ring for emission of fuel, deflects fuel from the downstream orifices. A flameholder grid in the duct is mounted for rotation about an axis transverse to the duct.

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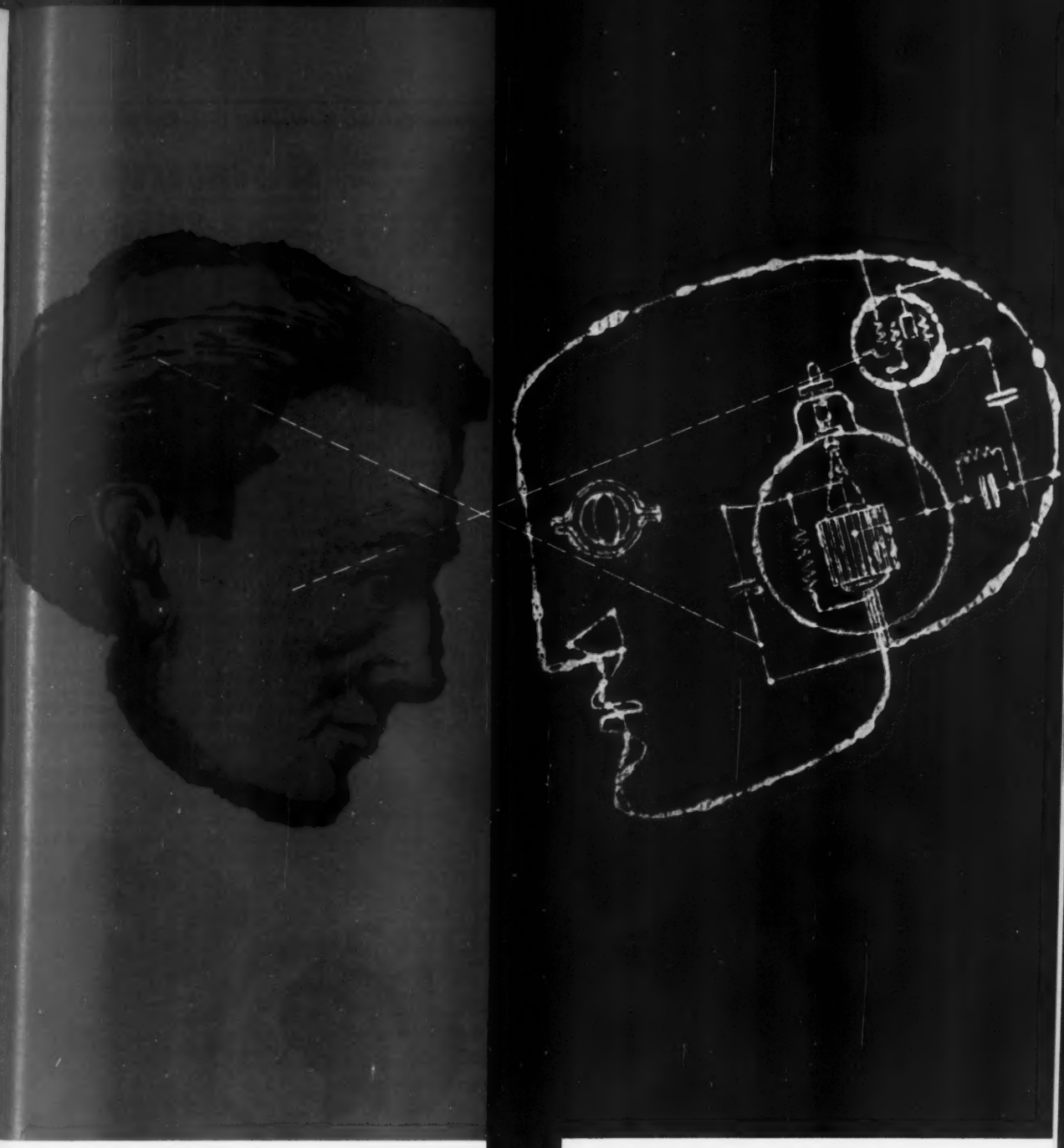

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Book Reviews

Ali Bulent Cambel, Northwestern University, *Associate Editor*

Aircraft Gas Turbines, by C. W. Smith, John Wiley & Sons, New York, 1956, 448 pp. $9.

Gas Turbine Series, Vol. I. Cycles and Performance Estimation, by J. Hodge, Academic Press, 1955, 329 pp. $9.50.

Jet Propulsion—Turbojets, Revised Edition, by V. C. Finch, The National Press, 1955, 327 pp. $5.50.

Gas Turbines and Jet Propulsion, Sixth Edition, by G. G. Smith, revised and enlarged by F. C. Sheffield, Philosophical Library, 1955, 412 pp. $15.

Reviewed by JOHN F. LEE
North Carolina State College

When one considers the paucity of permanent literature which existed in the gas turbine field less than eight years ago it is gratifying to see the appearance of four quite different volumes in such rapid succession. The volumes by C. W. Smith and J. Hodge are new contributions to the literature. The other two volumes are revised versions of earlier editions with little change in perspective or format. None of the volumes is a textbook in the usual sense, although "Aircraft Gas Turbines" could serve as an excellent text when supplemented by problem material and additional notes. All of the volumes, with the exception of the one by Hodge, are devoted primarily to aircraft gas turbines. There are other significant differences. The absence of illustrative material in the two volumes by C. W. Smith and J. Hodge presents an interesting contrast to the other two volumes which rely heavily on photographs and drawings. In fact, the authors of the latter two volumes assume the role of reporter of the gas turbine scene. For this reason, these two volumes are likely to have less permanent value.

"Aircraft Gas Turbines" by C. W. Smith is a serious and important contribution to the gas-turbine literature. The author has used his industrial and teaching experience to skillfully organize the material in a clear and logical pattern. All equations are completely developed and in every instance the basic assumptions are made clear. The introduction of the isentropic factor is especially commendable in view of the resulting simplification of equations and the greater ease of computation. Although there is not a single photograph of an existing engine this book is ample refutation of the ancient Chinese proverb.

"Cycles and Performance Estimation" by J. Hodge is a concise summary of working equations and cycle performance curves. Since this book is one of eight projected volumes in a series devoted to various aspects of the gas turbine, it is disappointing that a more inspired presentation of engine analysis was not given. After all, the engine process is not actually cyclic and hence the analyses of cycles depend on certain assumptions which the author rarely specifies. If the reader is sufficiently familiar with the assumptions and derivations which he must supply to obtain the working equations given in the book, it is not likely that he will have any need for this volume. The major value of the book is that of a handy reference in permanent form for the specialist.

"Jet Propulsion—Turbojets" by V. C. Finch is not much different from the earlier edition except in the addition of descriptions of newer engines and some later factual information. This volume can best be described as a book on computational procedures. The basic principles governing such computations are given short shrift. This criticism is not intended to belittle the real value of a knowledge of such procedures. A reader who has studied the pertinent principles in Smith's "Aircraft Gas Turbines" would do well to examine the computational outlines presented in Finch's volume. Indeed, if such procedures were included in "Aircraft Gas Turbines" the value of the book would be greatly enhanced. The historical notes and system descriptions in Finch's volume are grossly inferior to those

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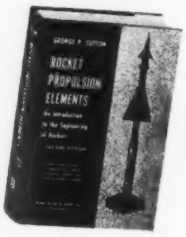

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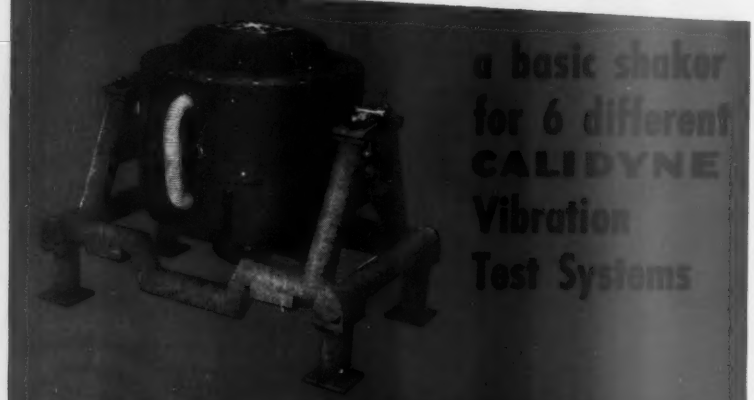

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"Gas Turbines and Jet Propulsion" by G. G. Smith is an excellent historical account of the development of the aircraft gas turbine. The emphasis is on the British contribution to the development. This is understandable since the author is British and it must be admitted that the British have undoubtedly made the greatest contribution in this field. However, the contributions of other nations are by no means neglected. This volume also describes in detail most of the important gas-turbine engines in existence in the Western World. As a source of historical fact and engine details this book is without a peer.

Men, Rockets and Space Rats, by Lloyd Mallan, Julian Messner, New York, 1955, 335 pp. $5.95.

Reviewed by ANDREW G. HALEY
AMERICAN ROCKET SOCIETY

Any person who desires to keep advised of the basic facts of the work being done during this advent of the age of space travel must read "Men, Rockets and Space Rats." The contents of this book are rare indeed, because only genuine, contemporaneous source material is presented. Thousands upon thousands of facts have been gathered, and have been presented with unembellished clarity. It is not recountal of past history. It is a record of the present. And so, while this book is of absorbing interest to the rocket men of today, it will also be a required work on source material for the author who, fifty years from now, is writing on the advent of the age of space travel.

The source material I refer to is really "source material" because it is not based upon hearsay, or on the writings of others, or on any other second-hand factors. The author traveled more than 18,000 miles, personally viewing and studying all of the facilities and installations concerning which he writes. And he personally interviewed the hundreds of persons involved. To insure complete accuracy at the source, he carried with him dictaphone equipment upon which he recorded each important conversation. He constantly referred to these recordings while writing the book and he checked and rechecked the statements of fact in the book with the direct recorded comments of the persons interviewed. Never before in history have the basic facts underlying the development of a great new industry been accumulated with such precision and on such a first-hand basis.

The book is replete with references to the incidents in the personal lives of the hundreds of scientists, engineers, and other technologists interviewed and studied, but the hard core of the author's effort is in no way injured by such personal anecdotes, as each anecdote is related to the evolution of some achievement or the solution of some problem. These precise etchings of human activity enliven the book and enlighten the reader and the value of the book is immensely enhanced thereby. This capturing of the basic on-the-scene realities of great episodes during the threshold of the space age constitutes, of course, the very marrow of true source material.

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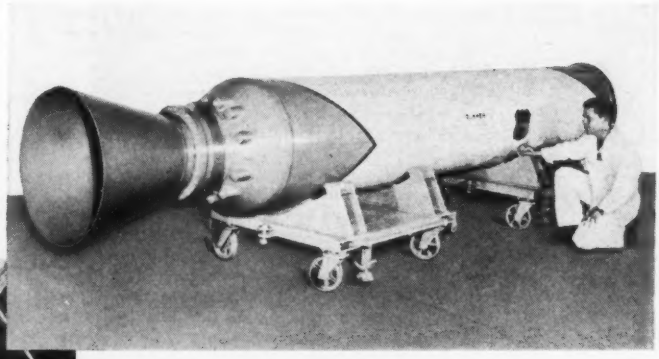

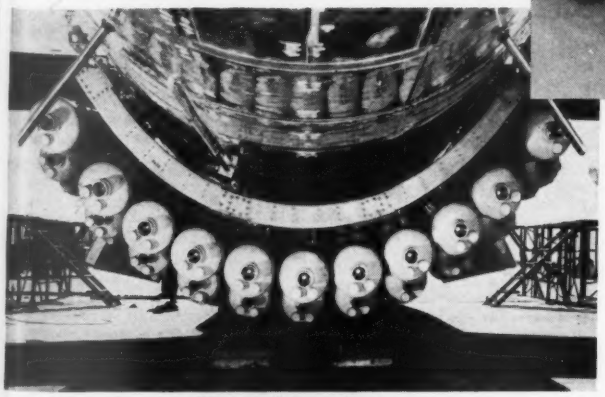

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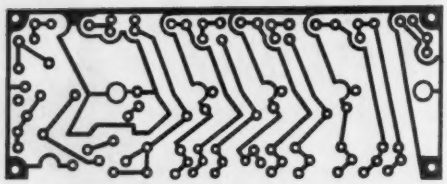

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If you have had engineering experience in any of the categories shown below, Northrop Aircraft has an attractive position for you, with many benefits. Important among them are high compensation, challenging assignments, steady advancement, recognition of initiative and ability, and continued interest in your progress. Many outstanding engineering positions are offered, as follows:

ELECTRICAL GROUP, which is responsible for the design of such things as power generation and distribution systems, rectifiers and power converters, and auxiliary systems as applied to manned aircraft, guided missiles and ground support equipment.

COMMUNICATIONS AND NAVIGATION GROUP, which is responsible for the design of C/N systems in manned aircraft and installation of guidance systems in missiles.

FIRE CONTROL RADAR GROUP, which is responsible for the installation and application of the most advanced type of fire control systems in fighter-interceptor aircraft. The work covers the installation of the equipment and associated wiring; continuing liaison with equipment manufacturers; preparation of system analysis and reports; and follow-up of system performance in the field as aircraft become operational.

INSTRUMENT GROUP, which is responsible for the design of instrument systems for manned aircraft and the installation of flight test instrumentation for guided missiles.

There are also opportunities for draftsmen with either electrical or mechanical experience.

At Northrop Aircraft you will be with a company that has pioneered for seventeen years in missile research and development. Here you can apply your skill and ability on top level projects such as Northrop's new supersonic trainer airplane, Snark SM-62 intercontinental missile, and constantly new projects. And you'll be located in Northrop's soon to be completed multi-million-dollar engineering and science building, today's finest in comfortable surroundings and newest scientific equipment.

If you qualify for any of these representative positions, we invite you to contact the Manager of Engineering Industrial Relations, Northrop Aircraft, Inc., ORegon 8-9111, Extension 1893, or write to: 1015 East Broadway, Department 4600- K, Hawthorne, California.

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NORTHROP AIRCRAFT, INC., HAWTHORNE, CALIFORNIA

Producers of Scorpion F-89 Interceptors and Snark SM-62 Intercontinental Missiles

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that the book is merely a series of disjointed essays. This is certainly not the case. The unities are preserved by very logical flowing of one subject into another, in a very satisfying procession of technological problems and solutions. The chronology of events is carefully followed.

We are first introduced to the men of the rocket age—most of whom are introduced with precision and with highly sensitive understanding of their achievements. There are some omissions—such as of Jim Russell, now Chief of the Bureau of Aeronautics, and A. M. O. Smith, now at Douglas—but these omissions are surprisingly few. From the old founders, the author whisks the reader to a synoptic look at a great proving center, namely, White Sands Proving Ground. From personalities he then introduces us to facilities and thereby sets the stage to understanding the rest of the book. At the same time, the author carefully introduces more and more people at the scenes of their technological efforts. He goes on to cover, in an interesting but relentless manner, field after field of rocket technology. At all times he is interesting and at all times he is factual.

He concludes the book with an absorbing seminar on the prospects of space flight. The concluding chapter includes verbatim interviews with such distinguished workers as Boushey, Stapp, Truax, Randall, Carter, Murray, Yeager, Bridgeman, Crossfield, Francisco, Henry, Tombaugh, Stein, Haas, Everest, Gompertz, Evans, Moore, Ridley, and Demming. I repeat these names only because, to any student of the field, they immediately reflect a cross section of the most highly qualified experts whom any researcher might interview.

Book Notices

Selected Combustion Problems, II. Butterworths Scientific Publications, London; Interscience Publishers, New York, 1956, 495 pp. $12.50. This volume contains the papers (and their discussions) presented at the AGARD Combustion Colloquim held in Liege, Belgium, on Dec. 5–9, 1955. A prepublication review of this book was carried in this section last July (JET PROPULSION, vol. 26, pp. 604–606). This volume is invaluable to jet propulsion scientists who are concerned with transport phenomena, ignition, altitude behavior, and scaling of aeroengines.

Proceedings of the Ninth Annual Conference on the Administration of Research, New York University Press, New York, 1956, 107 pp. $4. This volume constitutes the proceedings of the conference mentioned in the title and held at Northwestern University, Sept. 7–9, 1955. The subjects covered are: research relationships among government, industry and universities; industry and universities as sources of research for government; the future of research.

Elementary Differential Equations, by W. T. Martin and E. Reissner, Addison-Wesley Publishing, Reading, Mass., 1956, 260 pp. An introductory book on differential equations for students of science and engineering. This book is written with a modern viewpoint and should enable the reader to develop confidence in mathematical analyses.

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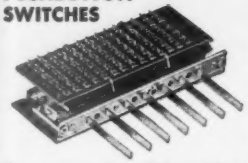

Two types; 1 to 24 pushbuttons, and up to 32 contacts per button.

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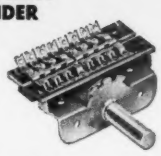

Type 160 gives a maximum of switching in space 7/8" high.

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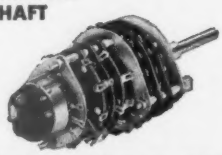
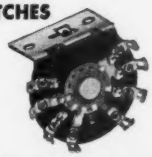


Combination has two switch assemblies, concentric shafts, AC switch.

PLUG SWITCHES



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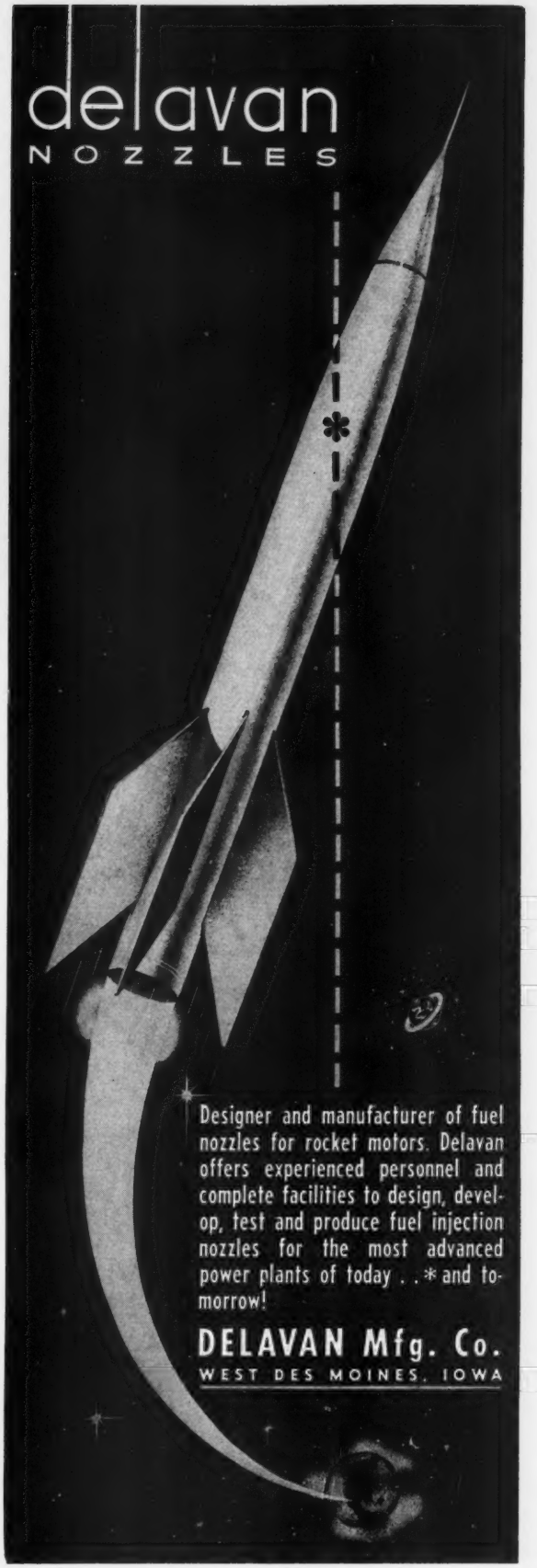

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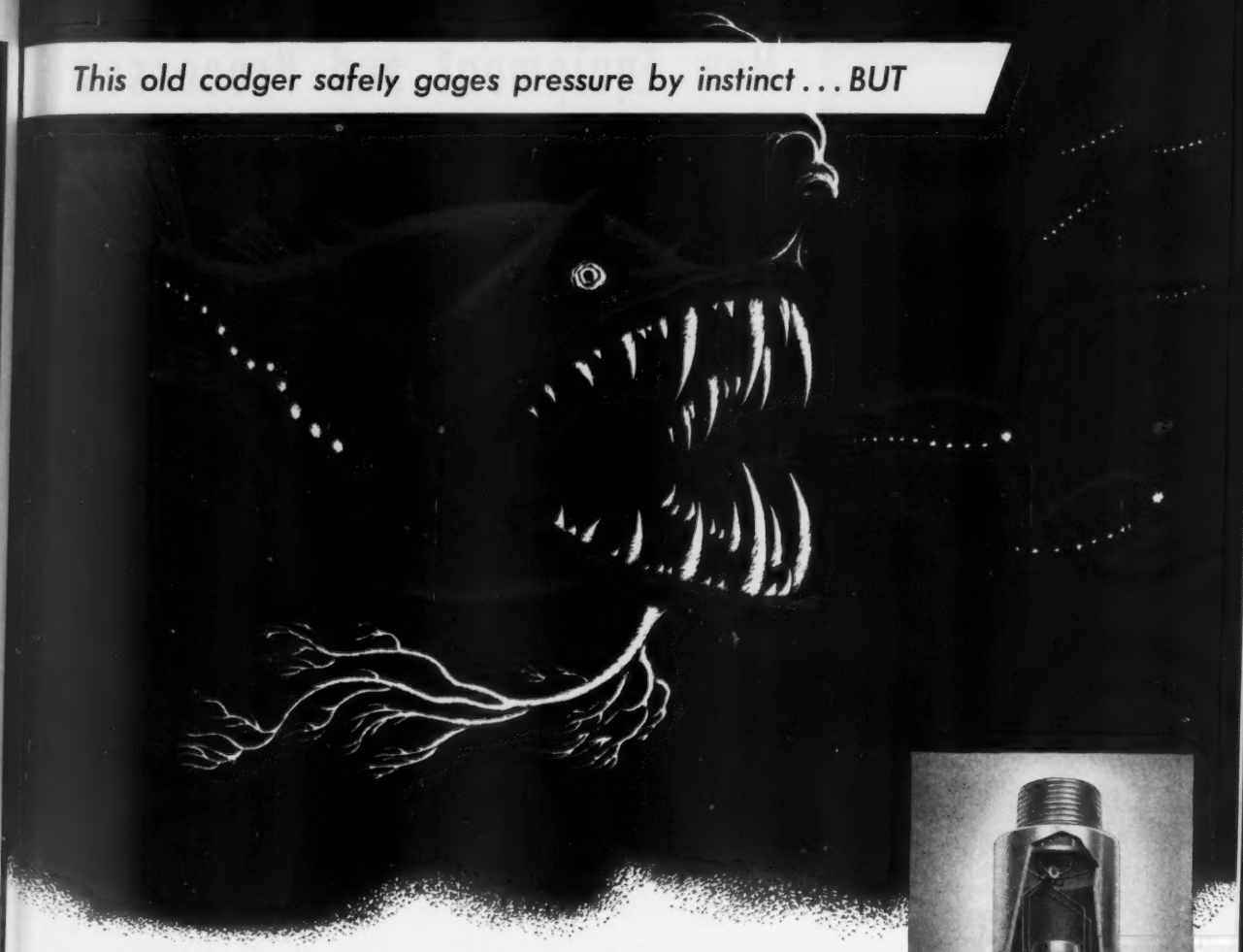

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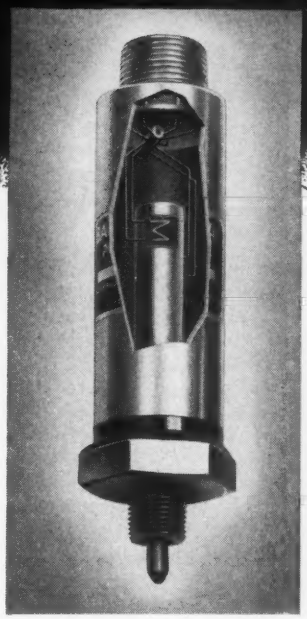
B-L-H Fluid Pressure Cells are *the one way* to measure absolute or differential pressures with consistent accuracy to within ±¼%. They convert pressure changes directly into electrical changes —no moving parts to wear out, no long pressure lines with possible leakage and fire hazards. The signal can be fed to Baldwin indicators, recorders, controllers or data processing instruments—for our fishy friend, it could even activate a servomotor that would angle his fins downward and take him out of danger.

Whatever your pressure measurement problem, a B-L-H representative will be happy to help you, from selecting the proper transducer to engineering a complete pressure measuring system. Remember, a system is only as accurate as the transducer. Write today for your free copy of Bulletin 4306 on rugged, compact, reliable SR-4 Fluid Pressure Cells.



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Missile Wiring. Multibranch, multiconductor breakout cable; 141 conductors. Pacific Automation Products, Inc., 1000 Air Way, Glendale 1, Calif.

Tantalum Capicator. Size "M" is 0.095 in. diam by 11/64 in. long. Capacitance of 0.02 to 4 mfd. Ohmite Mfg. Co., 3639 Howard St., Skokie, Ill.

Ceramic Tetrode. Stacked type is rated at 300 w, has high reliability, operates at 250 C. Eitel-McCullough, Inc., San Bruno, Calif.

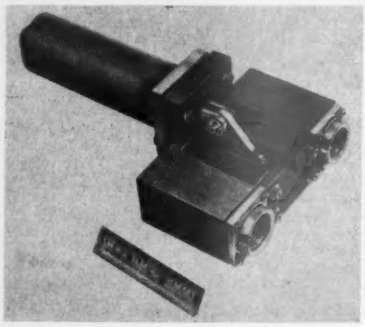
Missile Connector. Printed circuit type has 15 coded contacts mounted in plastic. Electronics Div., DeJur-Amsco, 45-01 Northern Blvd., Long Island City 1, N. Y.

Power Pack. Envelopes electric motor, hydraulic pump, valves and reservoir. For guided missiles, supersonic aircraft. Adel Precision Products, Burbank, Calif.

Subminiatureer Blower. Model BC-910B-1 has 1-in.-diam motor and weighs 5 1/4 oz. Operates at 115 v, 400 cycle ac at 20,000 rpm with 1250 hp. Delivers 8 cfm at 1-in. back pressure. Induction Motors Corp. 570 Main St., Westbury, L. I., N. Y.

Mechanical

Propellant Extruder. NRM 1-in. heavy duty model for single or double-base propellant; 3/4 hp motor. Notional Rubber Machinery Co., Clifton, N. J.



Propellant Valve. Straight-thru design for 0–1500 psi operation. Models for LOX, JP-4, JP-5, air, helium, RFNA, WFNA, hydrazine, nitrogen, aniline, ethylene oxide, propyl nitrate, hydrogen peroxide, ammonia, water. Hydromatics Inc., Cedar Grove, N. J.

Fuel Control Unit. For use on turbine-type auxiliary power units. Single or dual valve systems control JP-4, FNA. Flows of 500–15,000 lb/hr. Pierce Governor Co., 1625 Ohio Ave., Anderson, Ind.

Midget Packing Gland. Model MPG-AL features 24S-T4 with bores of 1/16, 3/32, 1/8, and 3/16 in. Temp range of −300 to 1850 F, pressure 500–7500 psi. Also in stainless steel. Conax Corp., 7811 Sheridan Drive, Buffalo 21, N. Y.

Thru-Tapped Cap Nuts. Self-locking eliminates washers, uses Nylon pellet. Nylok Corp., 611 Industrial Ave., Paramus, N. J.

High Flow Regulator. Inlet pressure of 6000 psi. Burst max is 15,000 psi. 1/4-in. size with internal filter. Accessory Products Corp., Dept. JP, 616 W. Whittier, Whittier, Calif.

Metering Valve. Series "A" in 3/8- and 1/2-in. size with 5/16-in. orifice, requires torque of 24 in.-lb to operate at 6000 psi. Teflon, Nylon, or Formica seats. Robbins Aviation, 1735 West Florence, Los Angeles 47, Calif.

Hydraulic Piston Pump. For pressures to 6000 psi, exceeds 90 hp. Handles up to 4.15 gpm. Dynex Inc., 1500 S. Muskego Ave., Milwaukee, Wisc.

Missile Valve. Solenoid operated. Open or closed position. For LOX, acids, etc. Clary Corp., San Gabriel, Calif.

Vacuum Pressure Switch. GAB 1000 series designed to respond to 2-in. Hg absolute. Resists self-actuation under vibration to 2000 cps at 10 g. Weighs 10 oz, is 3 1/2 in. long. Gorn Aircraft Controls Co., 845 Main St., Stamford, Conn.

Pressure Seal. Designed for shafts, Hexseal Series N-9000 permits intermittent speeds to 3600 rpm. Operation from −120 to 500 F. Shaft sizes of 1/8, 1/4, 3/8, and 1/2 in. Automatic & Precision Mfg. Co., 252 Hawthorne Ave., Yonkers, N. Y.

Rigid Plastic Valves. Polyvinyl for temp to 170 F and pressures to 170 psi. Kel-F needle. Chemtrol Corp., 11008 Santa Fe Ave., Lynwood, Calif.

Oxidant Valves. Series 5000 are designed for LOX, FNA, or helium. Teflon seal, 320–450 F fluid temp; 0-5000 psi. Precision Equipment Co., 1641 Border Ave., Torrance, Calif.

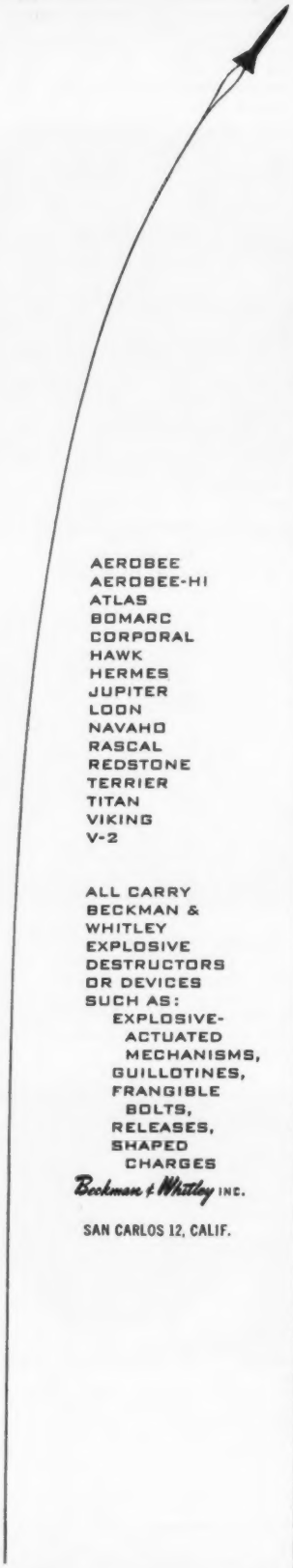
Check Valve. Controllable for 3000 psi, Series 5900 is zero-leak operation, 2–8 psi cracking pressure. 90 degree shaft rotation. Tactair Valve Div., Aircraft Products Co., 300 Church Rd., Bridgeport, Pa.

PVC Check Valve. Internals of polyvinyl chloride with seals of PVC, Neoprene Buna N with Nylon reinforcing. Stainless steel hardware. Techno Corp., 16 West 5 St., Erie, Pa.

Test

Direct Writing Oscillograph. Visicorder records 2000 cps. 6 channels on 6-in. chart. Galvanometer deflection is 6 in. peak-to-peak. 100 ft of record. Chart speeds of 0.2, 1, 5, and 25 in. per sec, min, or hr. Heiland Div., Minneapolis-Honeywell Regulator Co., 5200 E. Evans, Denver 22, Colo.

Pressure Pickup. Type 4-340 uses polar liquid and porous disk. Will respond flat between 3-2500 cps. Features high stability, high signal output. Consolidated Electrodynamics Corp., 300 N. Sierra Madre Villa, Pasadena, Calif.



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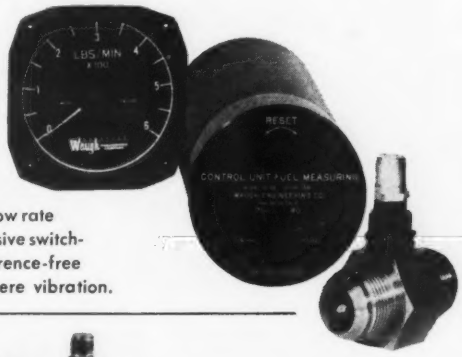
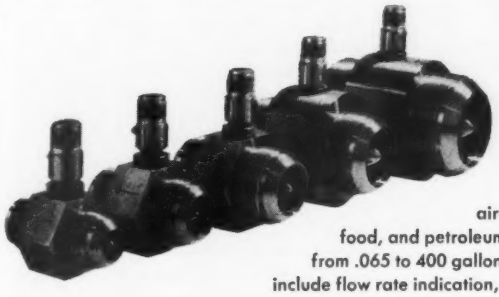
New, completely transistorized Airborne Fuel Measuring System with specific gravity adjustment provides direct indication of both fuel flow rate and total fuel consumed. Exclusive switch-type flow pickup gives interference-free operation in conditions of severe vibration.





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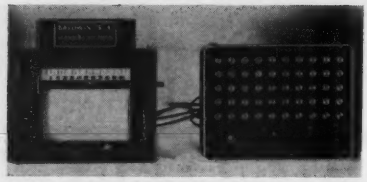

Strain Gage Scanner. New model allows 50-point automatic scanning and recording for SR-4 strain gages. Handles 2- or 4-arm bridges. Electronics Div., Baldwin-Lima-Hamilton Corp., 806 Massachusetts Ave., Cambridge 39, Mass.

Electrolytic Hygrometer. Provides 0–1000 ppm water determinations with ±5% accuracy. Sample flow of 100 cc per min. Beckman Instruments, Inc., 2500 Fullerton Rd., Fullerton, Calif.

Streamlined CRT. Type 401 offers calibrated sweeps, redesigned control panel. Allen B. DuMont Laboratories, Inc., 760 Bloomfield Ave., Clifton, N. J.

Portable X-Ray. 130 kv model is portable, weighs 50 lb and can penetrate 1¼-in. steel. Holger Andreason, Inc., Dept. 130, 730 Market St., San Francisco, Calif.

Pyrometer Kit. Includes calibrated meter, 6 ft of thermocouple wire, meter mount, all materials, and instructions. Two models: Chromel-Alumel with range of 0-2300 F, and Iron-Constantin to 1600 F. Pacific Transducer Corp., 11836 W. Pico Blvd., Los Angeles 64, Calif.

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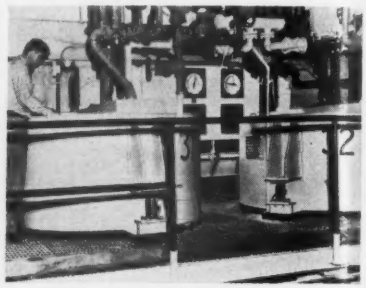
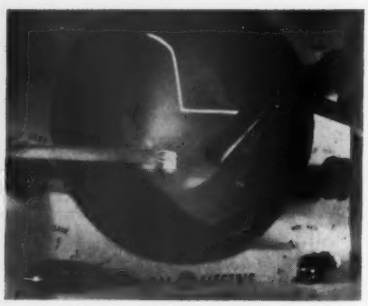
Ultra-Thin Strip. Close tolerances in many metals are now possible in strip widths to 4 in. and thicknesses from 0.01 in. down to 0.00012 in., with tolerances of 0.00005 in. Produced in a miniature integrated steel mill, the company will roll strip from metal furnished by customer or will produce metal to fit specs. Typical uses: diaphragms, springs, tape-wound cores or sheaths. Allied Products Div., Hamilton Watch Co., Lancaster, Pa.



In Process Weighing. Placed under tank supports, SR-4 load cells are being used to weigh additions to processes. Mixing by weight, rather than volume, is possible without any process interruption. The load cells are completely sealed. Baldwin-Lima-Hamilton Corp., Electronics & Instrumentation Div., 806 Massachusetts Ave., Cambridge 39, Mass.

Rokide Coatings. Demonstration of spray coating process displays ease with which these new coatings can be applied to a variety of base materials to protect against high temperatures, abrasion, erosion, and corrosion. Norton Co., Worcester 6, Mass.



High Temperature Electronics. Diodes operating at 800 C, triodes at 600 C, amplifiers at 1500 F, transformers and motors at 500 C may make possible high temperature components for the ICBM. Many of these new items have also been successfully run under high neutron fluxes and high temperatures and will find applications in the aircraft nuclear propulsion field. General Electric Research Lab., Schenectady, N. Y.

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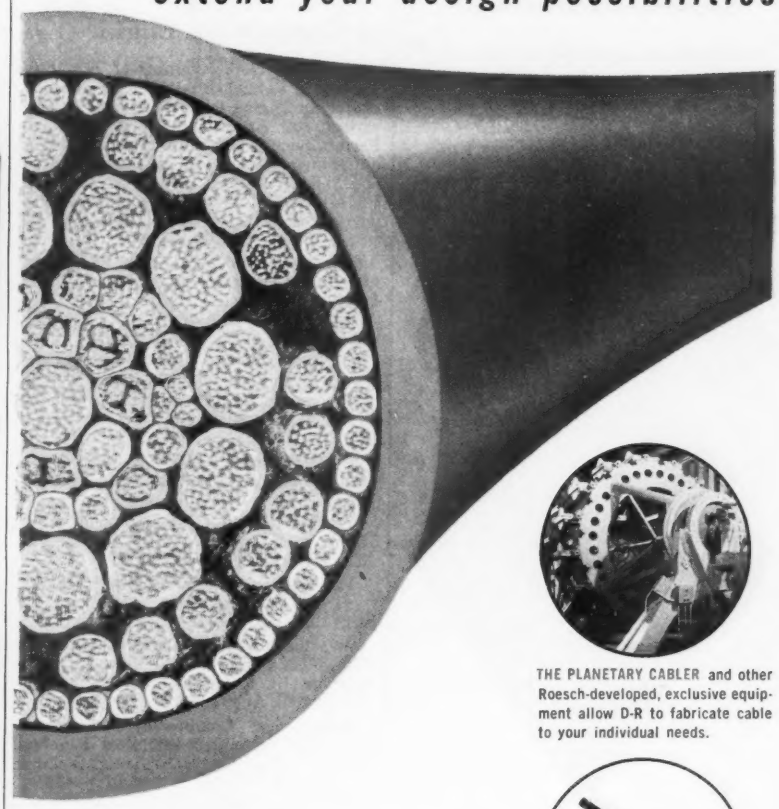

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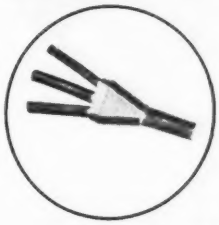

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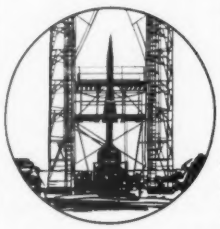

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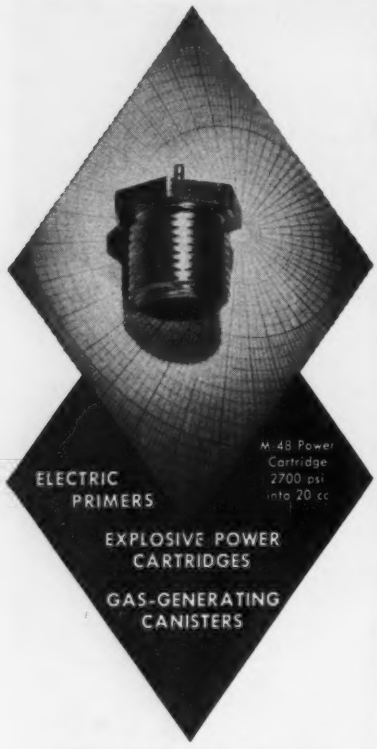

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Jet Propulsion Engines

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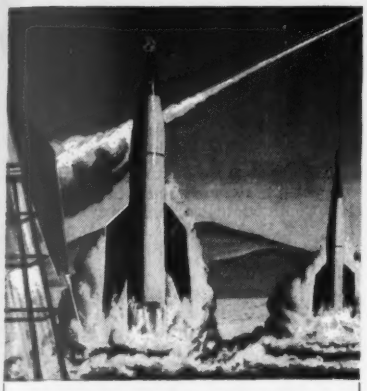

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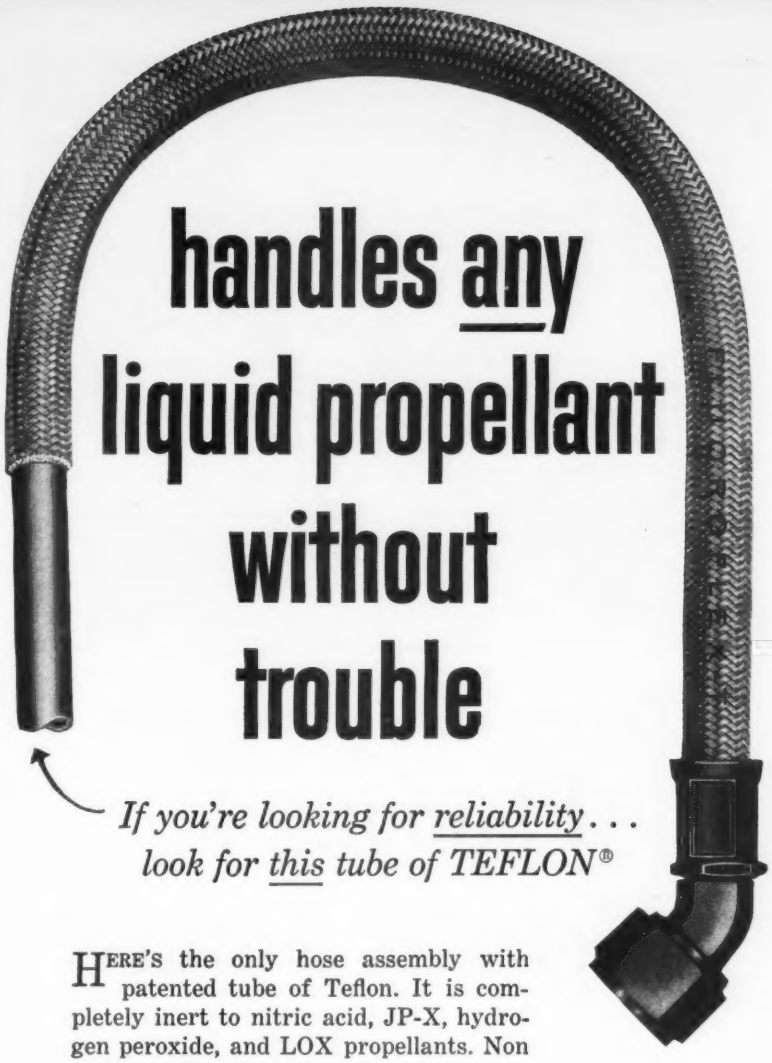

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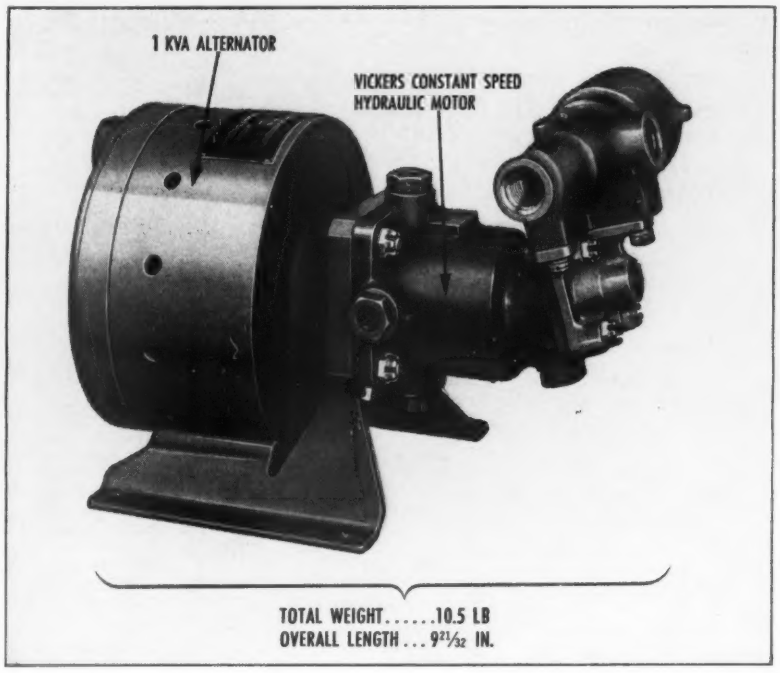

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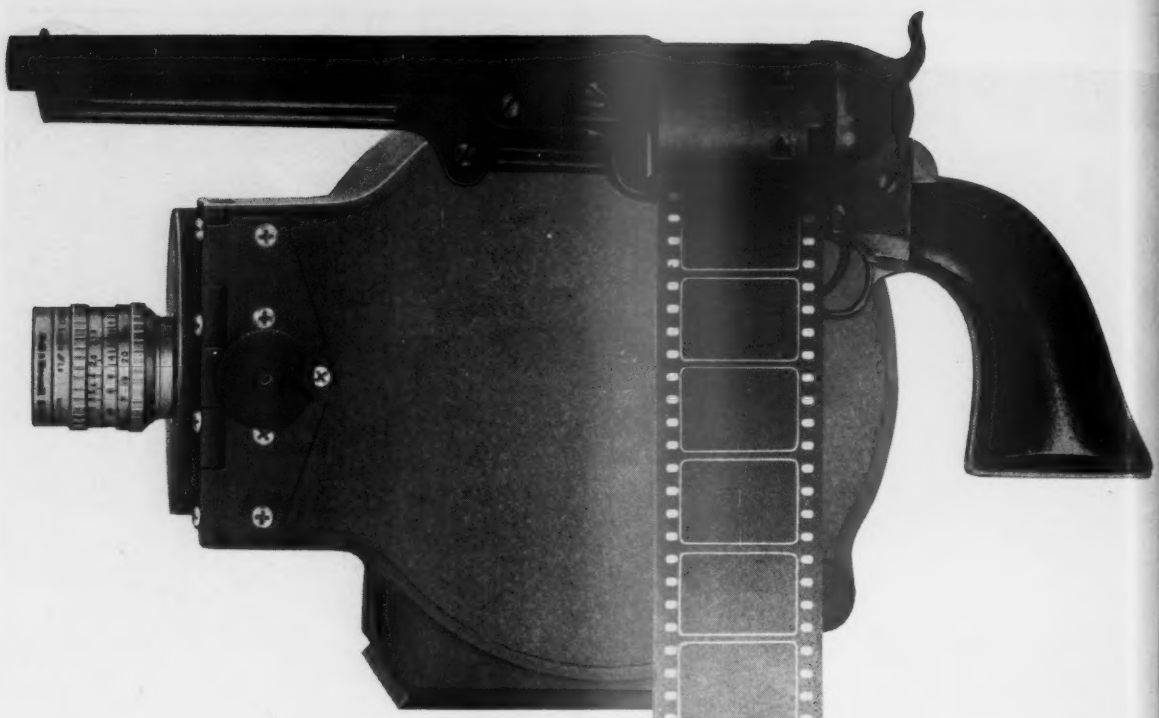

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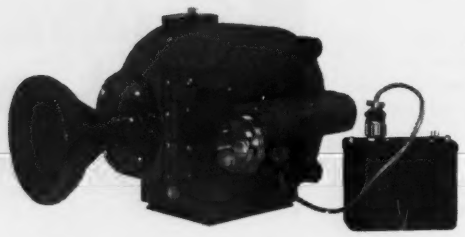

TRAID 500—Intermittent, 200 foot capacity, 16 mm. Tolerance of 30 G's, altitude 50,000 feet, temperature of −65° to +150°F. 6½" x 7⅝" x 5¼". Lens: Pacific Optical 3.45 mm f/1.5 with iris, 160° vertical and horizontal angle.

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